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VOLUME 2 - DETAILED TECHNICAL REPORT

BOOK 1

1 SEPTEMBER 1965

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VOLUME 2

DETAILED TECHNICAL REPORT

BOOK 1

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Apollo Support Department
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PREFACE

This document is the final detailed technical report of analyses and mission descriptions developed by General Electric Company under Contract NASw-410, for missions within the National Aeronautics and Space Administration program of near-earth space and lunar exploration. This report was prepared in accordance with the work statement, paragraph B.4.3.4, and NASA Headquarters Technical Direction Memorandum TD B4-2.

This book contains Part I, Sections 1 through 5. Book 2 contains Sections 6 and 7 and all of Part II.

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INTRODUCTION

The Apollo Support Department of the General Electric Company has conducted flight mission analyses based on NASA-furnished configuration, experiment, and flight mission assignments. This final report documents the results of the analyses, which included the preparation of detailed mission profiles for a series of Alternate Apollo Missions, the conduct of system performance analyses, and parametric mission sensitivity studies. The final report is submitted in four volumes: Volume 1 - Summary; Volume 2 - Detailed Technical Report, Books 1 and 2; Volume 3 - Appendixes; and Volume 4 - Mission Planning Aids.

Part I of Volume 2 defines seven mission profiles for missions scheduled in the NASA program of lunar exploration and scientific investigation of near-earth space. The seven missions, summarized briefly in Tables 1 and 2, are the following:

Mission 1 - Intermediate Inclination Survey.
Mission 2 - Saturn IB to Saturn IB Rendezvous.
Mission 3 - Synchronous Observation Post.
Mission 4 - Near Polar Retrograde Survey.
Mission 5 - Saturn B to Saturn IB Rendezvous.
Mission 6 - LEM Shelter/Taxi.
Mission 7 - Lunar Polar Orbit Survey.

The mission profiles were constructed for each mission from the NASA-furnished documents listed in the bibliography. Each profile includes the following elements:

a. A mission summary.
b. A statement of mission objectives, ground rules, constraints, and assumptions.
c. A description of the flight configuration and weights.
d. A description of the experiment packages aboard the vehicle.
e. Mission event profiles and operational descriptions of spacecraft systems and crew activities.
f. Configuration and weight histories.
g. A crew task schedule.
h. Tracking, communication, and experiment data profiles.

i. The trajectory description and ground trace.

j. An abort capability analysis.

Several detailed studies of various aspects of the Alternate Apollo Missions were accomplished in support of profile definition. The results of the following five studies are reported in Volume 2, Part II.

a. Environmental Analysis.

b. Lunar Site Photographic Coverage Analysis.

c. Data Flow Analysis.

d. Safety Hazards Analysis.

e. Rendezvous Launch Window Analysis.

Additional review tasks are also reported in Volume 2, Part II. Technical material of a reference nature is included in Volume 3, Appendixes. This material was directly derived from NASA-furnished source documents which are liberally referenced where used.

a. Experiment Constraints.

b. Lunar Site Locations.

c. AMPTF Crew task adaptation for operational tasks.

d. Detailed Crew Task Descriptions.

(1) Operational Tasks.

(2) Experiment Tasks.

A detailed evaluation of the input data appears in each mission profile (Part I, Sections 1 through 7). A summary of the problem areas which includes the requirements for additional data and the evaluation of data incompatibility, appears in Volume 1, Summary.

As a supplement to the final detailed report, Volume 4, Mission Planning Aids, is submitted. This document is intended to aid in the further detailed planning of the earth orbital missions.

Bibliography references in this report are given in parentheses following the data presented. In an example reference (see 9-3), the first number refers to the document as listed in the bibliography, and the second number refers to the page number of the document.

Table 1

AAM Earth Orbit Missions (a)

Mission Number	1	2		3	4	5	
Mission Designation (a)	Intermediate Inclination Survey	SA-IB to SA-IB Rendezvous		Synchronous Observation Post	Near-Polar Retrograde Survey	SA-V to SA-IB Rendezvous	
Launch Vehicle	SA-IB	1st Launch SA-IB 2nd Launch SA-IB		SA-V	SA-V	1st Launch SA-V 2nd Launch SA-IB	
Mission Orbit Requirements							
Altitude (n mi.)	200	200		19,350	200	200	
Inclination (degrees)	50	29.5 (48)		0	96.5	29.5 (48)	
Eccentricity	0 (Circular)	0 (Circular)		0 (Circular)	0 (Circular)	0 (Circular)	
Duration (days)	14	45 per spacecraft · 90 total		45	45	45 per spacecraft: 90 total	
Launch Data		1st SA-IB	2nd SA-IB			SA-V	SA-IB
Flight Assignment Number (46)	215	218	219	516	518	523	229
Launch Date (mo/yr) (1)	5/69	12/69	1/70	6/70	10/70	8/71	9/71
Launch Pad (7)	37B	34	37B	39A	39A	39B	37B
Crew Size (48)	3	3	3	3	3	3	3
Launch Geometry							
Launch Operations Description	2 Stages to 80-n. mi. Parking Orbit	2 Stages to 80-n. mi. Parking Orbit	2 Stages to 80-n mi. Parking Orbit	3 Stages to 100-n mi. Parking Orbit	3 Stages to Mission Orbit, Yaw Steering During S-IC and S-II Burns	3 Stages to 200-n. mi. Mission Orbit	2 Stages to 80-n. mi. Parking Orbit
AMR Launch Azimuth (degrees)	44	83 (C)	Variable	90	146	83 (C)	Variable
Parking Orbit Description							
Altitude (n. mi.)	80	80	80	100	Not Applicable	Not Applicable	80
Inclination (degrees)	50	29.5 (48)	Variable	28.5			Variable
Eccentricity	0 (Circular)	0 (Circular)	0 (Circular)	0 (Circular)			0 (Circular)
Insertion By	S-IVB	S-IVB	S-IVB	S-IVB			S-IVB
1st Transfer Trajectory							
Trajectory Type	Hohmann	Hohmann	Hohmann	Hohmann	Not Applicable	Not Applicable	Hohmann
Perigee Altitude (n. mi.)	80	80	80	100			80
Apogee Altitude (n. mi.)	200	200	200	19,350			200
Plane Change (degrees)	0	0	0	2 (48)			0
Inclination (degrees)	50 degrees	Coplanar	Coplanar	26.5 (48)			Coplanar
Insertion By	SPS	SPS	SPS	S-IVB			SPS
2nd Transfer Trajectory	Not Applicable	Not Applicable	Not Applicable	Elliptic	Not Applicable	Not Applicable	Not Applicable
Trajectory Type							
Perigee Altitude (n. mi.)				5,294 (48)			
Apogee Altitude (n. mi.)				19,350			
Plane Change (degrees)				17.4 (48)			
Inclination (degrees)				9.1 (48)			
Insertion By				S-IVB			
Mission Orbit							
Insertion By	SPS	SPS	SPS	SPS	S-IVB	S-II	SPS
Plane Change	0	0	0	9.1 (48)	0	0	0
Max. Wt. in Orbit (lbs)	30,929	33,185	33,185	61,441	122,974	210,667	33,185
Deorbit Planning Data							
Retro Fuel (lbs) (b)	794	810	824	14,554	861	884	824
Returnable S/C Deboost Weight (lbs)	20,664	21,057	21,421	35,119	22,432	21,421	21,421
Re-Entry							
Vacuum Perigee	0 (48)	0 (48)	0 (48)	26 ± 14 (29)	0 (48)	0 (48)	0 (48)
Recovery Site (24)	PRA (24)	PRA (24)	PRA (24)	PRA (24)	PRA (b)	PRA (24)	PRA (24)

Footnotes: (a) All information is from reference No. 19 unless otherwise noted.
(b) Retro Velocity impulse plus 10%.
(c) See Part I, Paragraph 2.2.5.

Table 2
AAM Lunar Missions (a)

Mission Number	6		7
Mission Designation	LEM Shelter/Taxi Mission		Lunar Polar Orbit Survey
	Shelter Delivery	Taxi/Exploration	
Launch Vehicles	SA-V	SA-V	SA-V
Launch Data			
Flight Assignment No. (46)	519	520	522
Launch Date (mo/yr) (1)	12/70	2/71	6/71
Launch Pad (7)	39B	39A	39A
Crew Size (48)	3	3	3
Launch Geometry			
Launch Operation Description	3 Stages to Earth Parking Orbit	3 Stages to Earth Parking Orbit	3 Stages to Earth Parking Orbit
Launch Azimuth (deg)	90	90	90
Earth Parking Orbit Description (24)			
Altitude (n. mi)	100	100	100
Inclination (deg)	28 5	28.5	28.5
Inserted by	S-IVB	S-IVB	S-IVB
Translunar Trajectory			
Type (47)	Retrograde Non-free Return	Retrograde Non-free Return	Retrograde Non-free Return
Maximum Flight Time (hrs) (24)	110	110	110
Maximum Midcourse Corrections (24)	3	3	3
Maximum Injected Weight (lbs) (47)	95,000	95,000	95,000
Inserted by (24)	S-IVB	S-IVB	S-IVB
Lunar Parking Orbit			
Altitude (n. mi.) (47)	80	80	80
Inclination (deg)	Variable (48)	Variable (48)	90 (46)
Eccentricity (47)	0 (Circular)	0 (Circular)	0 (Circular)
No. of Plane Changes	0 (24)	0 (48)	0-3 (48)
Duration (days)	90 (48)	14 (46)	28 (46)
Inserted by (24)	SPS	SPS	SPS
Descent Trajectory			Not Applicable
Trajectory Type (24)	Hohmann	Hohmann	
Perigee Altitude (feet) (24)	50,000	50,000	
Apogee Altitude (n mi.) (24)	80	80	
Inclination (deg) (48)	( )	( )	
Inserted by (24)	LEM Descent Engine	LEM Descent Engine	
Powered Descent			Not Applicable
Type	Automatic (47)	Automatic to Hover (24) Manual to Touchdown	
Landing Site	25° 20' E/1° 20' S (48)	25° 20' E/1° 20' S (48)	
Lunar Surface Operations			Not Applicable
Duration (days)	90 (1)	14 (46)	
Lunar Launch	Not Applicable		Not Applicable
Type		Automatic	
Plane Change Requirement		( ) (48)	
Parking Orbit	Not Applicable		Not Applicable
Altitude (feet) (24)		50,000	
Inclination (deg)		Coplanar with CSM	
Insertion by		LEM Ascent Engine	
Rendezvous Ascent Trajectory (24)	Not Applicable		Not Applicable
Type		Hohmann	
Perigee Altitude (feet)		50,000	
Apogee Altitude (n. mi.)		80	
Inclination (deg)		Coplanar with CSM	
Inserted by		LEM Ascent Engine	
Rendezvous and Docking Operations	Not Applicable		Not Applicable
Type (24)		Nominal Apollo	
Transearth Trajectory			
Maximum Flight Time (hrs) (24)	110	110	110
Maximum Midcourse Corrections (24)	3	3	3
Vacuum perigee (n. mi.) (29)	26 ± 14	26 ± 14	26 ± 14
Re-entry			
Recovery Site (24)	PRA	PRA	PRA

Footnote: (a) All information supplied by technical direction unless otherwise referenced.

SECTION 1

MISSION 1

INTERMEDIATE INCLINATION SURVEY

1.1 MISSION 1 DESCRIPTION

1.1.1 GENERAL

The Intermediate Inclination Survey Mission will map the visible and near-visible spectra regions of the earth. A spacecraft, consisting of a basic Apollo Block II Command Service Module (CSM) and a modified Lunar Excursion Module (LEM), manned by a crew of three will be placed in a nominal 200-n. mi. circular earth orbit at a 50-degree inclination. The nominal mission duration is 14 days (see 19).

1.1.2 MISSION SUMMARY

1.1.2.1 General

The spacecraft will be launched by the Saturn IB launch vehicle into a nominal 80 n. mi., circular, earth-parking orbit, with a 50-degree inclination (see 19).

During the parking-orbit phase of the mission, transposition and docking of the CSM and LEM is accomplished, then the S-IVB, including the Instrument Unit (IU) and the Spacecraft LEM Adapter (SLA), is staged (see 24).

During the second parking orbit, the SPS is used to insert the spacecraft into a co-planar Hohmann transfer orbit with an apogee of 200 n. mi. The spacecraft coasts in the transfer orbit until apogee, at which time the SPS is fired a second time, to circularize orbit at the nominal 200-n. mi. mission altitude (see 2).

The interlock and LEM cabin is then pressurized and the LEM is manned. Upon completion of the required LEM subsystem checks, the mission experiments are initiated, and the astronauts begin a work/rest cycle. The crewmen perform, adjust, and calibrate any experiments and associated apparatus, perform their station-keeping duties, and accomplish their normal spacecraft functions throughout the nominal 14 days of the mission.

1.1.2.2 Configuration

1.1.2.2.1 Launch Vehicle

The launch vehicle used for this mission is the Saturn IB, which consists of two propulsive stages (the S-IB and S-IVB) and a nonpropulsive module designated as the Instrument Unit (see 101-6).

1.1.2.2.2 Spacecraft

The spacecraft for this mission is the Apollo Block II CSM, LEM, a Launch Escape System (LES), and a Spacecraft LEM Adapter capable of supporting missions of 14 days duration (see 101-2).

1.1.2.3 Experiments

The experiments planned for this mission concern the biomedical and behavioral characteristics of crew members, artificial gravity producing equipment, radiation monitoring and techniques, and equipment for mapping and obtaining multispectral signatures of minerals and vegetable/animal life (see 35-2).

The total weight of the equipment required for these experiments is approximately 3300 pounds. The equipment will occupy 224 cubic feet, and require an average power of 703 watts, with peaks to 1298 watts* (Table 1-2).

1.1.2.4 Operations

The spacecraft is launched from Cape Kennedy Launch Complex 37B, using the Saturn IB vehicle. The launch is on an azimuth of 44 degrees, into the 80-n. mi. parking orbit at an inclination of 50 degrees (see 19).

After the parking orbit is attained, the crew performs subsystems checks and accomplishes orbital landmark navigational sightings. The orbit ephemeris is established and verified by the Manned Space Flight Network (MSFN).

During the parking orbit phase of the mission, transposition and docking is accomplished, and - shortly after a hard dock is made between the CSM and the LEM - the S-IVB,

*Power requirements for the artificial-g experiments (0301), which are not defined at this time, must be added to these totals.

including the IU and the SLA, is staged. The final portion of the parking orbit phase is used for preparations for SPS transfer orbit insertion.

Upon completion of transfer orbit insertion preparations, the SPS inserts Vehicle A into a co-planar Hohmann transfer orbit with an apogee of 200 n. mi. and Vehicle A coasts in the transfer orbit until apogee, at which time the SPS is fired a second time in order to place the spacecraft in a circular orbit at the nominal 200-n. mi. mission altitude. After the post-insertion tasks are completed, the interlock and LEM are pressurized and manned. Upon completion of the required LEM subsystem checks, the mission's prime experiments are initiated.

The astronauts begin a work/rest cycle. The crewmen perform, adjust, and calibrate any experiments or associated apparatus, perform station-keeping duties, and accomplish their normal spacecraft functions for the duration of the mission.

Fourteen days after launch, the vehicle de-orbits, re-enters, and is recovered.

1.1.3 MISSION OBJECTIVES

1.1.3.1 Experiment Objectives

The basic objectives of the experiments to be performed during this mission (see 35-2) are:

a. Qualifications of remote sensing equipment and demonstrations of manned survey techniques for mapping and obtaining multispectral signatures of minerals and vegetable/animal life.
b. Observations in agriculture, forestry, geology, oceanography, and of geography groundtruth sites and uncharted regions.
c. Prototype feasibility testing of artificial gravity experiments.

The experiments are as follows (see 35-2):

a. 0100 Biomedical.
b. 0200 Behavioral.
c. 0301 Artificial-g.
d. 0501 Radiation Environment Monitoring.
e. 0901 Multispectral Target Characteristics.
f. 0902 Synoptic Earth Mapping.

1.1.3.2 Flight Systems Objectives

1.1.3.2.1 Launch Vehicle

The launch vehicle will place the spacecraft payload into an 80-n. mi. parking orbit, where the CSM can transpose and dock; by means of the SPS, the spacecraft will effect a Hohmann transfer to the operational altitude of 200 n. mi. at a 50-degree inclination (see 19).

1.1.3.2.2 Spacecraft

The spacecraft will carry the mission experiments and the crew of three to the specified mission orbit, return the crew and data samples, and provide operational support for the crew and experiments throughout the 14-day mission (see 1).

1.1.3.2.3 Crew

The crew, consisting of three astronauts (see 9-4), will be expected to participate in the performance of experiments beyond merely checking and operating equipment. The crew will help to ensure the scientific and technical validity of experiments and provide partial evaluations of results (see 13).

1.1.3.3 Ground Support Objectives

1.1.3.3.1 Mission Command Control

The mission command responsibility will remain the same as for the basic Apollo mission, with the ability to pass command to the onboard crew commander for all but the critical mission phases (see 15).

1.1.3.3.2 Launch Support and Control

The Launch Control Center and the normal complement of equipment at Pad 37B (see 7) will be utilized for the mission launching. Modifications to launch facilities and to checkout operations will be minimized (see 9-13).

1.1.3.3.3 Tracking and Communications

All MSFN remote sites supporting this mission will communicate directly with Mission Control Center (MCC) in Houston, with sufficient data routed in real time to preclude the necessity for remote site controllers (see 15-11). Tracking and communications will be required continuously throughout the Saturn IB powered flight (see 15-66).

1.1.3.3.4 Recovery

The mission recovery operations will be similar to the basic Apollo system, with the Pacific recovery range as the primary recovery area (see 24). Potential use of recovery/rescue aircraft will be considered in lieu of continuous assignment of ships for world-wide coverage of abort conditions (see 9-70).

1.1.3.3.5 Post-Flight

The objectives for post-flight operations, involving such activities as data reduction, equipment performance review, crew performance, and ground support performance reviews, were not considered in the program documentation of the Alternate Apollo Mission.

1.1.4 MISSION GROUND RULES

1.1.4.1 Earth Launch Phase

Mission 1 will be launched from Launch Complex 37B of the Merritt Island Launch Areas (MILA) (see 1-31).

The launch azimuth for this mission will be 44 degrees (see 49). The minimum operational earth launch window will be 2-1/2 hours (see 24-4).

Continuous tracking is required from launch to three minutes after the insertion of the vehicle into the initial orbit (see 91).

1.1.4.2 Earth Parking Orbit

The nominal earth parking orbit will be a near-circular orbit of 80 n. mi. altitude (see 24-4).

Transfer orbit injection will occur not later than three orbits after injection into the initial orbit (see 47-2). Injection into the transfer orbit will occur at the earliest opportunity after spacecraft systems have been checked and the orbit ephemeris has been determined (see 29-2). For initial studies, the third orbit will be selected for the nominal injection orbit (see 29-2).

There will be no natural lighting constraints on transposition and docking (see 29-3), other than the requirements for sun illumination of the LEM docking tunnel and the sun vector angle of less than 90 degrees to the CSM X-body axis.

Ground rules can not be established at this time on communications requirements during transposition and docking.

MSFN will be the primary source for ephemeris determination (see 26-5). A minimum of two station contacts is required during the first parking orbit; one station contact is required during all subsequent parking orbits (see 14-33). For station contact considerations a contact time of at least four minutes at an elevation angle of 5 degrees or higher is required (see 91).

1.1.4.3 Transfer Trajectory Phase

The transfer trajectory will be a Hohmann ellipse (see 19).

One four-minute pre-burn contact with the ground network within 90 minutes before burn is required (see 91). One four-minute post-burn contact starting no later than 7 minutes after burn is required (see 91).

1.1.4.4 Mission Orbit Phase

The nominal mission orbit will be a near-circular orbit of 200 n. mi. altitude with an inclination of 50 degrees (see 19).

The flight mission is under control of the flight director, who under normal circumstances will delegate the control of the scientific program after the spacecraft has achieved its design and mission orbit (see 14-55). One four-minute contact with the MSFN unified S-band system is required per orbit (see 91). The time between two successive contacts will not exceed two hours (see 91).

The design of the spacecraft will permit extra-vehicular transfer of the crew between the CSM and LEM in either direction in both the docked and undocked conditions (see 24-3). The SPS will be used for de-orbiting (see 14-63).

At least one astronaut in the spacecraft will be wearing a spacesuit at all times during flight. The faceplate may be open or closed (see 24-2). Not more than two astronauts will be off duty at any one time during the mission (see 76-5).

Ground rules for experiment scheduling are not available at this time. A series of assumptions were used to produce an example schedule, which is presented in paragraph 1.2.3.

1.1.4.5 Re-Entry Phase

The re-entry trajectory characteristics are as follows:

a. Vacuum perigee - 0 n. mi. (see 49-2).
b. Re-entry corridor - Not defined at this time (see 29-7).
c. Re-entry range - Not defined at this time (see 29-7).
d. Re-entry inclination limits - Not defined at this time.
e. Cross-range maneuvering capability - Not required with the nominal entry trajectory (see 24-5).

One four-minute pre-burn contact with the ground network within 90 minutes before retro is required. One four-minute post-burn contact is required not later than seven minutes after retro (see 91).

The primary recovery area will be the Pacific Recovery Area (see 29-9).

1.1.5 ALTERNATE APOLLO MISSION WEIGHTS SUMMARY

Table 1-1 summarizes the spacecraft weights for Mission 1. The table is broken into three sections: Command Module, Service Module, and LEM/LAB. For each module the major categories are listed and referenced.

The command module is an NAA Block II Modification (see 45). The service module is an NAA Block II Modification (see 45), except for RCS fuel loading, which is based on Grumman AES Studies (see 31); and SPS retrofuel load, which is based on computations done by General Electric, assuming impulse burns and an additional 10 percent ΔV margin included in the retrofuel budget (Part II, Section 6).

The LEM/LAB weights are based on GAEC experiment assignments (see 31) as altered by Bellcomm (see 35) to meet margin requirements.

Table 1-1

Mission 1 - Flight 215 Intermediate Inclination Survey
Spacecraft Mission Orbit Injection Weight Summary

Module		Weight Pounds		Reference No. and Page
1. Command Module				
Empty Weight		8,946		(45) No Page
Useful Weight		1,269		(45) No Page
	Total	[10,215]		
2. Service Module				
Empty Weight		8,388		(45) No Page
Reaction Control Propellant[1]		835		
Experiments RCS Propellant			(614)	(31) 1.5.15
Excess RCS Propellant			(221)	
Cryogenic Expendables		653		(45) No Page
SPS Retrofuel[2]		794		Part II, Section 6
	Total	[10,670]		
3. LEM/LAB				
Basic Structure		2,204		(31) 1.5.16
SCS		43		(31) 1.5.16
NGS		120		(31) 1.5.16
Crew Provisions		114		(31) 1.5.16
RCS		496		(31) 1.5.16
(Propellant)			(185)	(31) 1.5.16
Controls and Displays		330		(31) 1.5.16
Environment Control		579		(31) 1.5.16
Instrumentation		217		(31) 1.5.17
EPS		1,807		(31) 1.5.17
(Reactants and O_2)			(366)	(31) 1.5.17
Communications		99		(31) 1.5.17
Experiments		3,312		(31) 1.5.17
Spares		9		(31) 1.5.17
	Total	[9,330]		
4. Total Spacecraft Weight[3]		30,215		
Maximum Spacecraft Weight into Orbit[4]		30,929		(49) Table 10
Margin		714		

1. Total SM RCS Tank Capacity - 835 pounds.
2. This figure includes a 10 percent ΔV reserve.
3. Summation of all square bracketed numbers.
4. This value will increase (see 130); however, no data is available for this launch trajectory at the present time.

1.1.6 EXPERIMENT LOAD DESCRIPTION

1.1.6.1 Support

The experiment loads for Mission 1 are summarized in Table 1-2. The total weight of the equipment required for experiments is approximately 3,300 pounds, of which more than half is attributed to earth-mapping equipment.

The experiments planned for this mission include biomedical, behavioral, artifical-g, radiation environment monitoring, multispectral target characteristics, and synoptic earth mapping.

The ground-based operational support for the biomedical and behavioral experiments include periodic data reception, and monitoring of safety parameter data. In radiation environment monitoring, the external particle flux measurements made during solar flare events are correlated to measurements from earth observatories and earth satellites (see 31).

For the synoptic earth-mapping experiment, a network of ground trackers is required to establish (within 10 feet) the position of the orbiting spacecraft relative to reference points within the photographic field of view (see 31).

For the artificial-g experiment, ground-based operational support requirements include the monitoring of real-time commentary audio or teletype from the astronauts of the status of deployment, spin, de-spin, and other critical events of the experiment. Artificial-g will be simulated in the CM and LEM by using the LEM as a counter-weight in a rotating cable-connected system (see Figure 1-1). To simulate lunar surface conditions, the level of artificial-g to be evaluated will range from 1/6 g to 1 g by varying the rate and the radius of rotation (see 31).

Since the artificial-g experiment requires rotation of the spacecraft and the mapping experiments require a stable earth directed attitude, these experiments require careful exclusive scheduling. Also, the spinning effect under artificial-g could interfere with the performance of other experiments, particularly in the biomedical and behavioral group (see 31.15).

Table 1-2

Experiment Load Summary

(35-2) (35-2) (35-2) (35-3)

Experiment Number	Experiment Category/ Abbreviated Title	Total Wt. (Lbs.)	Volume ($Ft.^3$)	Volume Reference	Power (Watts)	
					Average	Peak
0100	Biomedical	313	21.5	(32-1-7.5 and 6)	100	
				(31-1-5.5)	100	50
0200	Behavioral	20	2.3	(32-1-7.5)	100	
0301	Artificial-g	351	48.0	(31-1-5.5)	*	
0501	Radiation Environment Monitoring	47	2.64	(31-1-5.5)	6.3	
0901	Multispectral Target Characteristics	953	80.64	(31-1-5.5)	790	210
0902	Synoptic Earth Mapping	1628	69.03	(31-1-5.4)	485	770
	Totals	3312	224.11		703	1298

*Power requirements for 0301 are not available at this time.

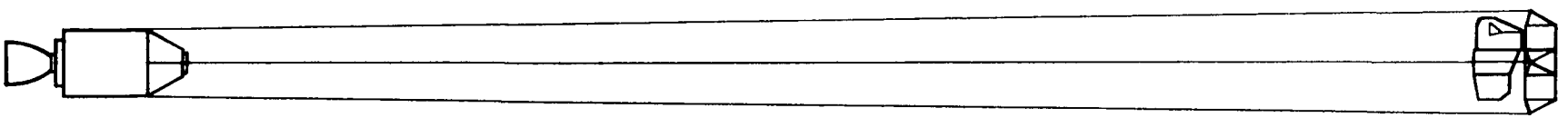


Figure 1-1. Mission 1 Artificial-g Experiment Configuration (See 31, Page 1-5.20)

Table 1-3

Mission 1 - Intermediate Inclination Survey

Phase	Initiating Event	Event Time Hrs Min Sec	Concluding Event	Duration ΔT Hrs Min Sec
1.0 PREFLIGHT	T -450 Hours	T -450h	Liftoff (T -0)	450h
2.0 EARTH ASCENT	Liftoff (T -0)	0.000	Parking Orbit Insertion-S-IVB Cutoff	595.54 sec[1]
2.1 S-IB Thrusting	Liftoff (T -0)	0.000	S-IB Staging	145.94 sec
2.1.1 Coast		141.64 sec[1]		4.3 sec
2.1.2 S-IB Staging		145.94 sec		
2.2 S-IVB Thrusting	S-IB Staging	145.94 sec[1]	Parking Orbit Insertion (S-IVB Cutoff)	449.60 sec
2.2.1 Ullage Case and Thermolag Jettison		155.94 sec[1]		
2.2.2 LES Jettison		165.94 sec[1]		
3.0 80-N. MI. EARTH PARKING ORBIT	Parking Orbit Insertion-S-IVB Cutoff	9 m 55.54 s	TOI-SPS Ignition	(3 h 10 m)[2]
3.1 Initial Coast	Parking Orbit Insertion-S-IVB Cutoff		CSM/LEM - S-IVB Separation	
3.2 Transposition and Docking	CSM/LEM - S-IVB Separation		Hard Dock	40 min
3.3 S-IVB/IU/SLA Staging	Hard Dock		TOI-SPS Ignition	
4.0 TRANSFER ORBIT COAST	TOI-SPS Ignition	3 h 20 m	EOI-SPS Cutoff	45 min
4.1 TOI-SPS Thrusting	TOI-SPS Ignition	3 h 20 m	TOI-SPS Cutoff	11.5 sec
4.2 Transfer Orbit Coast	TOI-SPS Cutoff		EOI-SPS Cutoff	44 m 37.35 s
4.3 EOI-SPS Thrusting	EOI-SPS Ignition	4 h 5 m	EOI-SPS Cutoff	11.2 s
5.0 200-N. MI. EARTH ORBIT COAST	EOI-SPS Cutoff	4 h 5 m	Beginning of De-Orbit Preparation	14 days
6.0 DE-ORBIT	Beginning of De-Orbit Preparations	14 days	Re-Entry	1 h 57 m
6.1 LEM Staging	Beginning of De-Orbit Preparations	14 days	LEM Jettison	1 h 30 m[3]
6.2 SPS Retrothrusting	LEM Jettison	14 d 1 h 30 m	SPS Retro Cutoff	12.5 sec
6.3 De-Orbit Coast	SPS Retro Cutoff	14 d 1 h 30 m	SM Staging Initiation	17 m [4]
6.4 SM Staging	SM Staging Initiation	14 d 1 h 30 m	Re-Entry	10 m [4]
7.0 RE-ENTRY	(400,000 Feet) Re-Entry	14 d 1 h 57 m	Drogue Chute Deployment	10.5 m[5]
8.0 PARACHUTE DEPLOYMENT	Drogue Chute Deployment	14 d 2 h 7 m	Touchdown	7 m[5]
9.0 RECOVERY RETRIEVAL	Touchdown	14 d 2 h 14 m	Retrieval	

1. Reference 49, Table 2.
2. Reference 26 Adapted for Daylight Docking.
3. Reference 26 Adapted from LEM Jettison in Lunar Orbit.
4. Section I, Paragraph 1.2.5
5. Reference 24. Page 4.13-3.

1.2 OPERATIONS

1.2.1 MISSION PHASE DESCRIPTION

1.2.1.1 Introduction

Mission 1, for the purposes of analysis and description, consists of nine major phases. Table 1-3 defines each major phase, its initiating event, the time of initiating event, the concluding event, and the time duration of each phase. The phase descriptions represent the detailed operations conducted and the crew activities performed during each phase.

Many parameters in the table do not currently have values assigned. These parameters were included, based on comparable work in the AMPTF Design Reference Mission, to establish the basis for the future development of mission phase descriptions. Trajectories are detailed in paragraph 1.2.5; experiment activities (crew functions) are detailed in Appendix D; the scheduling of experiments and flight crew operations are shown in paragraph 1.2.3. Crew activities adapted from AMPTF information are documented in Appendix C. The following is a detailed description of Mission 1 by phases.*

1.2.1.2 Phase 1: Preflight

The preflight phase begins with the securing of the launch vehicle to the launch pad, and ends with launch vehicle liftoff at T -0.

When the space vehicle is secured and all umbilical interfaces have been verified, the space vehicle subsystems are checked. The Guidance and Navigation System (GNS) is checked and provided with mission information. The liftoff time reference and real time data for vehicle performance monitoring are provided. The spacecraft is then fueled.

The LEM and CSM receive cryogenic servicing, cabin, crew and water supplies. The GNS of both the LEM and the CSM are checked out and activated. The CSM is cleaned and sterilized, and launch vehicle propellant loading is completed.

*This description is based on References 24, 26, and 29 and the summary chart of the AAM Earth Orbit Missions, Table 1, of this report.

The down range area is cleared, while tracking and communication networks receive final verification. Several hours before launch, the launch vehicle receives its cryogenic load. At t -6 seconds, the S-IB engine is ignited (the six seconds serve as a thrust buildup period). The vehicle is released and lifts off at T -0.

1.2.1.3 Phase 2: Earth-Ascent

The earth ascent phase begins with vehicle liftoff, at T -0, and terminates with S-IVB cutoff, after parking orbit insertion.

After liftoff, the S-IB rises vertically (pitch, 90 degrees) for 12 seconds and begins a preprogrammed pitch maneuver that smoothly reduces the pitch angle to zero degrees at S-IB burnout. The time and altitude of the highest axial acceleration are 140 seconds and 192,000 feet. At S-IB staging, the vehicle has a velocity of 6000 fps, at an altitude of 192,000 feet. The S-IB is staged prior to S-IVB ignition. S-IB engine cutoff and S-IB/S-IVB separation are reported to MCC by the commander.

During ascent, the navigator monitors the S-IB ignition, the caution and warning indicators, and the environmental control system panel. For the duration of the ascent phase, the engineer maintains radio communication with the control center, and monitors the critical temperature flow and event displays. The commander monitors acceleration, attitude, attitude rates, abort indicators, and pitch and roll rates.

S-IVB ignition occurs at T +145.9 seconds, and the thrust lasts for 449.6 seconds. The LES is jettisoned 20 seconds after S-IVB ignition. The vehicle pitch angle reduces from 38 degrees at S-IVB ignition to zero degrees at the 80-n. mi. earth orbit insertion point. At S-IVB cutoff, T +595 seconds, the vehicle's velocity is 25,665 fps and its altitude is 80 n. mi.

After verifying S-IVB ignition, the commander continues to monitor vehicle performance and confirms the LES jettison. Approximately 10 minutes after launch, the commander observes the S-IVB engine cutoff and vehicle insertion into the 80-n. mi. earth parking orbit.

1.2.1.4 Phase 3: 80-N. Mi. Earth Parking Orbit

The 80-n. mi. earth orbit phase begins with S-IVB cutoff and concludes with SPS ignition for transfer orbit insertion.

After S-IVB engine cutoff at the mission time of T +595 seconds, the commander and navigator confirm the achievement of the parking orbit. All three crewmen then adjust their couch restraints and set the controls and displays for parking orbit operations. Using the inflight test system, the navigator and engineer perform a checkout of spacecraft systems, including guidance and control, environmental control, reaction control, and electrical power. The engineer also continues to maintain communications with the MCC.

The commander next stabilizes the spacecraft attitude so that the navigator can perform the IMU fine alignment, and landmark navigational sightings. The S-IVB reaction control system (RCS) is used to orient the space vehicle for landmark acquisition.

Once he has determined and verified the orbit ephemeris, the commander makes the decision for mission continuation in the normal mode. The crew then prepares for transposition and docking maneuvers, orienting the spacecraft to the angle where the sun illuminates the LEM docking tunnel, and the sun vector makes an angle of less than 90 degrees with the CSM body X-axis. Translation and docking must occur within 4.5 hours after liftoff, the available lifetime of the S-IB-IU. After the S-IVB is placed in the attitude-hold mode, the CSM/S-IVB-LEM separation devices are activated.

When the CSM/S-IVB-LEM separation is assured, the spacecraft LEM adapter (SLA) is deployed to aid in the stabilization of the S-IVB-LEM and to expose the LEM. The CSM then translates 35 feet forward, performs a 180-degree pitch maneuver, and translates back to the S-IVB-LEM. The CSM performs docking maneuvers to align the CM-LEM interface and to engage the LEM drogue (X-axis docking tunnel). A maximum of 15 minutes is required from separation to the accomplishment of a soft dock.

After soft docking, the navigator pressurizes the LEM, then secures the docking latches by opening the CM hatches and fastening the hard dock latches. He then connects the umbilicals between the two modules. A maximum of 25 minutes is required from soft dock to the securing of the CM hatches after performing the hard dock.

Once the CM hatches are secured, preparations (including orientation) for S-IVB staging are completed, and the separation devices are activated, initiating S-IVB (including the IU and SLA/CSM-LEM) separation.

Upon completion of hard docking, the commander initiates the S-IVB separation from the CSM/LEM configuration. The separation is visually monitored by the engineer.

The commander then translates the CSM/LEM from the jettisoned S-IVB stage. The engineer reports the completion of the maneuver to MCC.

Following S-IVB staging, the crew prepares for insertion into the transfer orbit. Navigational sightings and IMU coarse and fine alignments are performed, and the decision is made to continue the nominal mission. The three crewmen adjust their seat restraints. After the GNS is programmed, the spacecraft is properly oriented and the countdown for transfer orbit insertion (TOI) is initiated. At a mission time of T +3 hours and 20 minutes, the GNS automatically commands SPS ignition. Five seconds prior to SPS ignition, SM RCS ullage ignition occurs.

1.2.1.5 Phase 4: Transfer Orbit

The transfer-orbit phase begins with the ignition of the SPS for TOI and terminates with SPS thrusting for earth orbit insertion.

During the second 80-n. mi. earth orbit, the SPS is used to transfer the spacecraft into a co-planar Hohmann transfer orbit with a nominal apogee of 200 n. mi. The geocentric latitude and longitude of the point of thrust initiation are 45.2 and -31.8 degrees, respectively. The navigator confirms the transfer orbit by computing the trajectory parameters and comparing these onboard calculations with values obtained from MSFN. The commander sets the controls for transfer orbit operations and checks the SPS parameters and propellant status.

During transfer orbit operations, the navigator performs a second IMU alignment in preparation for the second SPS burn into the 200-n. mi. orbit. During this period, the engineer checks the electrical power and service propulsion systems. The commander maneuvers the spacecraft for the IMU alignment and maintains communications with the MSFN. The three crewmen then monitor the SPS ignition, burn and cutoff.

Under GNS guidance, the SPS is used to insert the spacecraft in a circular orbit at the nominal 200-n. mi. mission altitude. This thrusting maneuver occurs at 136.6 degrees longitude, -45.2 degrees latitude, and T +4 hours and 5 minutes. The termination of EOI SPS thrusting ends the transfer orbit coast phase.

1.2.1.6 Phase 5: 200-N. MI. Earth Orbit Coast

The 200-n. mi. earth orbit coast phase begins with EOI SPS cutoff and terminates on the last mission day, 14 days from launch, when de-orbit preparations are begun.

The navigator confirms orbit injection using the spacecraft displays, the guidance computer, and data from MSFN. He then secures the controls and displays and assists the commander in a CSM system status check. While the engineer continues to record PCM telemetry and maintains ground communications, the navigator proceeds with the IMU fine alignment and the landmark navigation sightings. During these operations, the commander maneuvers the vehicle attitude for landmark acquisition. The navigator next compares his orbit information with ground-computed parameters to ensure the accuracy of the onboard calculations.

The engineer prepares to transfer to the LEM for earth orbit operations. The commander checks the LEM pressurization from the CM. Then, the engineer equalizes the CM and interlock pressure, removes and stows the CM pressure and thermal hatches, and disconnects and stows the probe and drogue. After activating and checking out his portable life support system, the engineer transfers from the CM to the LEM by proceeding through the tunnel. After an inspection of the LEM cabin, he closes the hatch and initiates communications with the MSFN. After a brief checkout period, experiments that require crew activation or participation are set up.

During each day of the orbit phase, the crew's primary activities include scientific experiments, system status checks, and navigational sightings. Personal functions are performed in the time remaining.

1.2.1.7 Phase 6: De-Orbit

The de-orbit phase begins when the crew starts de-orbit preparations, and terminates with CM entry into the earth's atmosphere at 400,000 feet.

At the end of the 14-day orbit phase of the mission, the engineer deactivates the LEM systems and transfers to the CSM. At this time, the crew transfers to the LEM all CM and experimental equipment that is to be jettisoned. Upon completion of these activities, each crew member enters his couch, fastens his restraints, and begins preparation for the LEM jettison.

After orienting the spacecraft for LEM jettison, the commander initiates LEM/CSM separation by igniting the shaped charges to sever the LEM/CSM interface. He then translates the CM from the LEM, using the CM RCS. The LEM is staged prior to the last orbit, at a mission time of 14 days. MSFN tracking should be available during LEM staging.

The commander maneuvers the spacecraft to the proper attitude, while the navigator performs a navigation fix. After completing the IMU fine alignment and a landmark navigational sighting, the navigator computes the onboard de-orbit trajectory. The MSFN supplies orbital parameters for comparison and the navigator's trajectory calculations are verified. The crew then checks all CM systems in preparation for the third SPS burn into the de-orbit coast phase.

The crewmen don their pressure suits and adjust their seat restraints. At the termination of the commander's countdown, the GNS automatically commands RCS ullage and SPS ignition. The navigator monitors the SM RCS ullage maneuver, and the commander and navigator monitor the SPS ignition, burn, and cutoff.

An SPS retrofiring of 12.5 seconds inserts the CSM into a descending transfer orbit with a vacuum perigee of 0 n. mi. The SPS thrust initiation begins nominally at a mission time of T +14 days, 1 hour, 30 minutes and at 31 degrees longitude and 0 degrees latitude.

With the termination of SPS retrothrusting, the commander checks the insertion paremeters, the CSM status, and communicates his findings to the MSFN. Next, he prepares for SM jettison by orienting the CSM to the proper attitude and by activating the CM-SM separation sequence. During this period, the engineer switches all systems from SM to CM, and retracts the antenna and all optics.

After final IMU alignment, the CSM is properly oriented and the SM staged. SM staging occurs 10 minutes prior to re-entry.

All three crewmen monitor the CM-SM separation. After SM separation, the navigator and engineer perform a post-separation status check of the CM RCS. The commander then maneuvers the CM to the MSFN confirmed entry attitude. Following a check by the commander of the entry monitor flight indicator, the crew members adjust restraints, check their suit umbilicals, and close their face plates.

Following SM jettison, a final pre-entry check is conducted, which includes checks of RCS and GNS entry, entry monitor systems, crew equipment systems, and CM orientation for re-entry. The de-orbit phase ends with the CM re-entry into the earth's atmosphere (a nominal altitude of 400,000 feet). Re-entry occurs at a geocentric longitude and latitude of 143.6 and -46.5 degrees in the Pacific Recovery area.

1.2.1.8 Phase 7: Re-Entry

The re-entry phase begins when the CM enters the earth's atmosphere, and ends at the time of drogue-chute deployment, which occurs at an altitude of 24,000 feet.

The re-entry phase begins at a mission time of T +14 days, 1 hour, and 57 minutes at the entry altitude of 400,000 feet, at a geocentric longitude and latitude of 143.6 and -46.5 degrees. The entry velocity is 25,831 fps, and the flight path angle, and azimuth are -1.52 and 67.6 degrees, respectively.

The initial part of the re-entry phase is a full-lift flight at 400,000 feet to a point where the rate of descent is reduced to 700 fps. At this point, trajectory range control is initiated. During trajectory range control, the steering commands are based on a reference trajectory, selected so that the guided trajectory predicted range, from the control initiation point to touchdown, matches the range requirements to reach the target. The final part of the entry phase consists of an equilibrum glide to an altitude of 24,000 feet, the altitude at which drogue-chute deployment occurs.

During re-entry, the commander continues to monitor the CM entry attitude and the glide maneuver. The navigator starts the elapse-event-time clock upon 0.05g indication. The engineer maintains communications with the MSFN recovery forces throughout this phase of the mission.

The CM earth landing system is armed during the equilibrium glide when the CM passes through an altitude of 100,000 feet. The forward heat shield is jettisoned at an altitude of 25,000 feet, about 1.5 seconds prior to drogue-chute deployment.

The navigator monitors the forward heat shield deployment at 25,000 feet, and the drogue-chute deployment 1.5 seconds later. If the parachute is not deployed automatically, he manually deploys the chute.

1.2.1.9 Phase 8: Parachute Descent

The parachute descent phase begins at an altitude of 24,000 feet, at the time of drogue-chute deployment, and ends approximately 7 minutes later with CM earth landing.

The CM at an altitude of 24,000 feet, is over the earth landing site location at aproximately T +14 days, 2 hours, and 7 minutes. The CM is at a geocentric longitude of

( ) degrees and a latitude of ( ) degrees at the time of drogue-chute deployment. The CM's relative velocity and flight path angle at this point is ( ) fps and ( ) degrees.*

The drogue chutes are jettisoned and main chutes are deployed 58 seconds after the beginning of the phase. Main chute deployment is observed by the crew four seconds below the 15,000-foot altitude at approximately 11,000 feet. Earth landing occurs at 420 seconds after the deployment of the drogue-chutes. At CM landing, the relative velocity and flight path angle are ( ) fps and -90 degrees. Earth landing occurs at a longitude and latitude of -170 degrees and -13 degrees.

Recovery communications (VHF) are activated and a final systems check is performed during parachute descent, starting 4 minutes and 49 seconds prior to touchdown.

After the navigator activates the VHF system, the three crewmen shut off all power switches not required for landing. The remaining propellant in the CM RCS is then dumped by the navigator. The commander continuously monitors the altitude and descent velocity during this phase, communicating the status of the descent to MSFN. Touchdown occurs approximately 10 minutes after heat shield deployment.

1.2.1.10 Phase 9: Recovery and Retrieval

The recovery and retrieval phase begins with CM touchdown and continues until both the CM and crew are onboard a recovery ship.

Immediately after touchdown, each crew member releases his seat restraints and assesses his physical condition and the structural condition of the CM. While the commander activates the post-landing recovery aids, the engineer checks out the EPS and ECS for post-landing operations. After obtaining survival equipment from the storage area, the crew egresses to await recovery.

*No printout of re-entry trajectory parameters is available.

1.2.2 CONFIGURATION

1.2.2.1 General

The command module is an NAA Block II Modification Configuration 1 (see 45).

The service module is an NAA Block II Modification 1 (see 45), except for the RCS fuel loading, which is based upon Grumman AES studies (see 31-1.5), and the SPS retrofuel loading, which is based on General Electric computations (Section 2, Part 6).

The subsystem modifications to the CSM are listed in Table 1-4 based on addition or deletion of items from a Block II CSM.

Table 1-4

Subsystem Modifications

Subsystem	Addition	Deletion
Communication and data	Wiring interface	High-gain and rendezvous transponders
ECS	Blower and ducts	
Electrical power	Wiring interfaces	

The LEM/LAB configuration and the location of experiment equipment for Mission 1 are shown in Figure 1-2. Conceptual sketches are available for several of the pieces of equipment used in the mission and are shown in Figure 1-3.

1.2.2.2 Experiment Equipment

The spacecraft configuration modifications are summarized by experiment location and major subsystem modifications (see 31-1.5).

A viewfinder installation provides the hemispherical coverage for all experiments. The optical sensor installation utilizes optics mounted in the LEM descent stage and on a dual-axis gimbal to accommodate the required higher pointing accuracy. During an extravehicular activity (EVA), the 0901-4 multispectral camera is replaced by the 0901-2A grating spectrometer sensor installation on the 0901-1 optics, and the film is changed. The infrared imager (0901-2B) is mounted in the descent stage. The

microwave imager antenna (0901-5B), is stowed in the descent stage and deployed for experimentation.

The microwave radiometer antenna (0902) is stowed in the ascent stage and is deployed for experimentation. The VHF pulse antenna (0902), stowed in the descent stage behind the microwave imager antenna (0901-5B), is deployed after the 0901-5B antenna. The 24-inch and 6-inch focal length cameras (0902) are mounted in the descent stage on EVA for film change.

The artificial-g test involves separating the CSM and LEM/LAB, retaining physical connection via a cable mechanism, and rotating the extended configuration about a common spin axis. The artificial-g cable mechanism is a four-cable system, using brake-equipped motorized winches mounted on the descent stage. The mechanism requires an astronaut to attach the cables to special fittings on the SM. Checkout of the mechanism and control concept will involve separating the CM and Lab to the maximum distance required for the artificial-g flight, and duplication of the rpm (g-levels) to be tested at the separation distance.

During this test one astronaut is in the CM and two astronauts are in the Lab. Attitude stabilization and control are available in both vehicles. For this experiment, the Lab carries the LEM flight control and RCS system.

Structural modifications to the LEM ascent stage are required to accomodate the viewfinder (0902). All other experiment equipment installations involve additions to existing structures of the LEM ascent and descent stages.

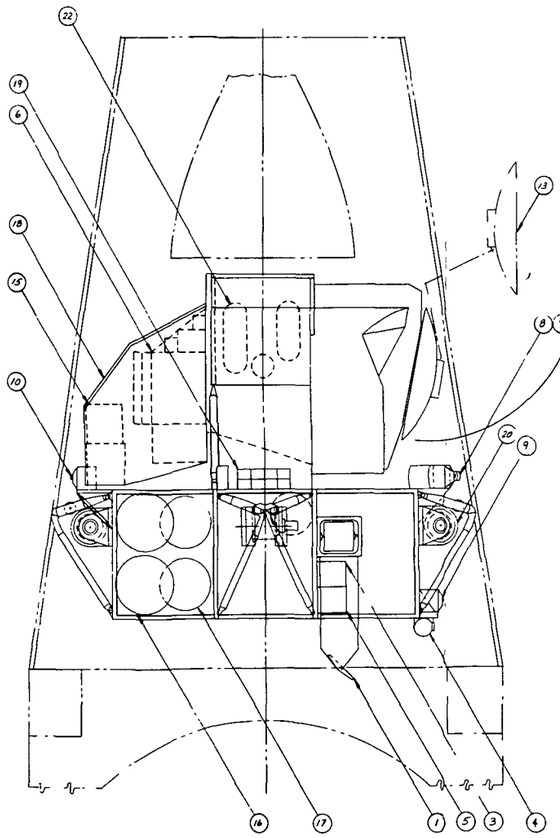
Adequate internal volume is available to accomodate experiments. All primary experiment displays and controls are located in the forward cabin or are accessible from forward cabin crew positions. Food is stored internally. A seven-day supply of lithium hydroxide canisters is stored also internally, mounted on the aft bulkhead of the Lab, and is replaced from external storage during programmed EVA's.

Supporting electronics for experiments are carried in the existing electronics bay mounted on the back of the ascent stage.

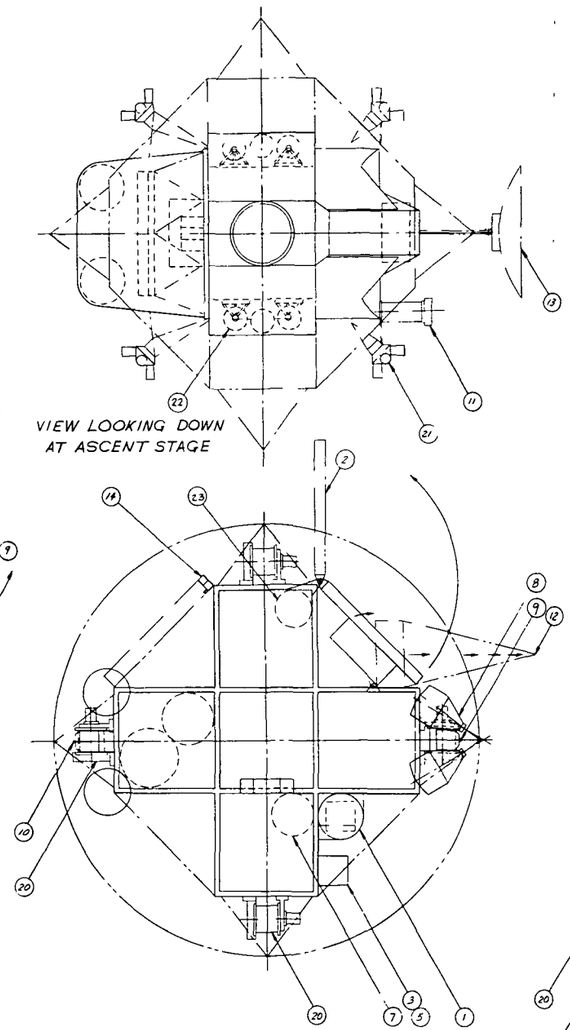
In addition to the standard CSM/LEM-LAB interface, this flight requires additional wiring to accommodate earth pointing requirements. The interface wiring involves star tracker and horizon scanner inputs, used to maintain an attitude hold.

KEY

No.	Experiment	Name
1	0901	16-in. Telescope
2	0901	Microwave Imager Antenna
3	0901	Grating Spectrometer
4	0901	Spectro Photometer
5	0901	Multi Spectral Camera
6	0902,0901	Electronics Bay
7	0901	Cryogenics Storage
8	0902	24-in. FL Camera
9	0902	6-in. FL Camera
10	0902	70mm Star Camera
11	0902	Image Viewfinder
12	0902	VHF Pulse Radar Antenna
13	0902	Microwave Radiometer Antenna
14	0501	Radiation Environment Spectrometers
15		Fuel Cells (2)
16		H_2 Tanks (2)
17		O_2 Tanks (2)
18		Radiators
19		LioH Storage
20	0301	Artificial "g" Winch (4)
21	0301	RCS
22	0301	RCS Fuel Tanks
23		GOX Tank

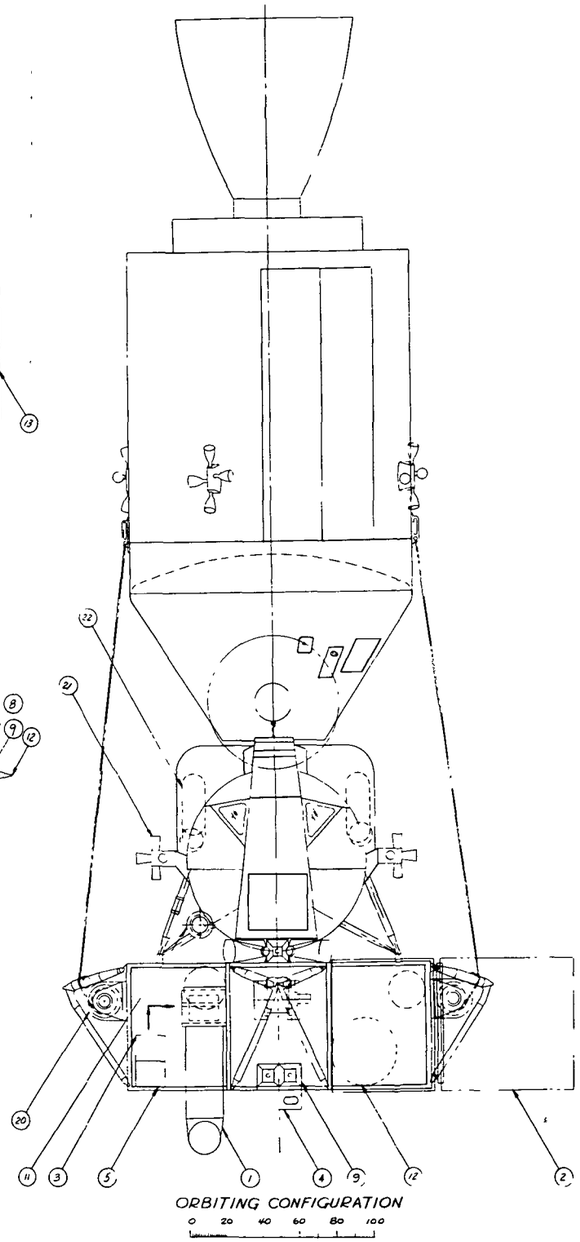


LAUNCH CONFIGURATION



VIEW LOOKING DOWN AT ASCENT STAGE

VIEW LOOKING DOWN AT DESCENT STAGE



ORBITING CONFIGURATION

0 20 40 60 80 100

Figure 1-2. Mission 1 Spacecraft Configuration and Experiment Location (31, Page 1-5.18) (Sheet 1 of 2)

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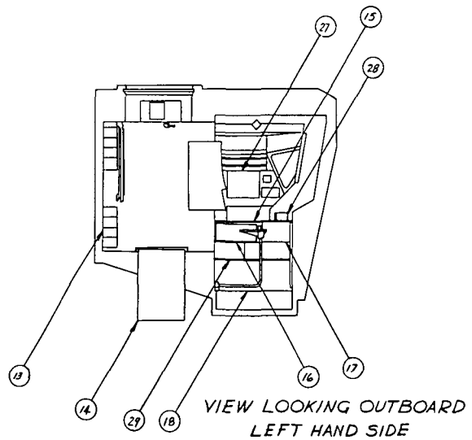
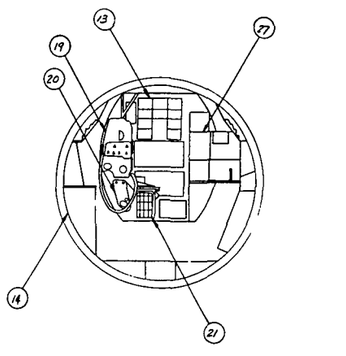
KEY

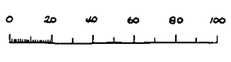
No.	Experiment	Name
1	0901	16-in. Telescope
2	0901	Spectro Photometer
3	0902,0901	Electronics Bay
4	0902	24-in. FL Camera
5	0902	6-in. FL Camera
6	0902	70mm Star Camera
7	0902	Image Viewfinder
8	0902	Microwave Radiometer Ant.
9		Fuel Cells (2)
10		H_2 Tanks (2)
11		O_2 Tanks (2)
12		Radiators
13		LioH Storage
14	0100	Biomed
15	0501	Pulse Height Analyser
16		Recorders
17		Digital Computer
18		Film & Tape Storage & Log Book
19		ECS
20		Water Module
21		Fan
22	0501	Battery Charger
23	0100-0200	Basic Lab
24		Main Panel Floodlights
25		Food Storage
26	0301	Artificial "g" Winch
27	0200	Integrated Human Performance Test Console
28	0501	Rate Meter & Ionization Chamber
29	0301	"g" sensor & Rotation Rate Sensor

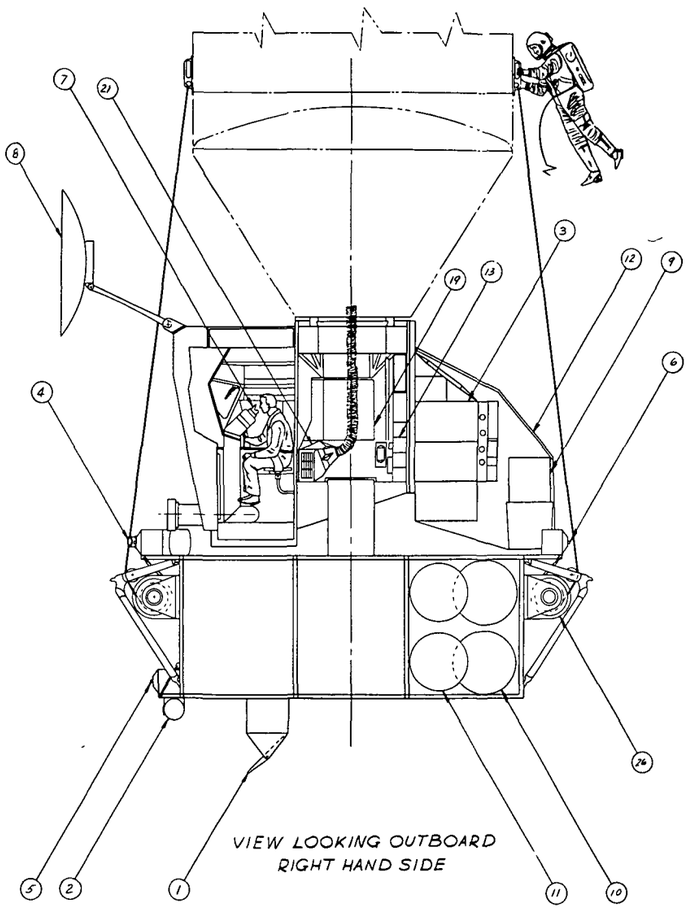


VIEW LOOKING OUTBOARD
LEFT HAND SIDE

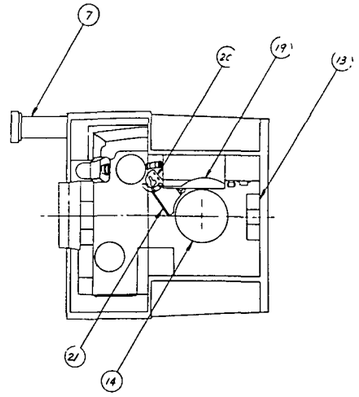


VIEW LOOKING AFT

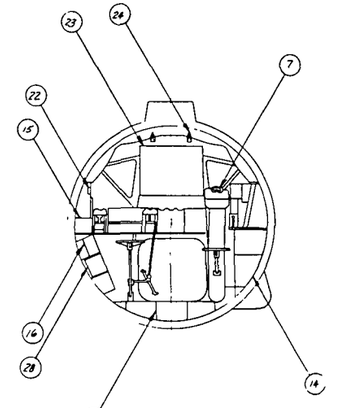




VIEW LOOKING OUTBOARD
RIGHT HAND SIDE



PLAN VIEW



VIEW LOOKING FORWARD

Figure 1-2. Mission 1 Spacecraft Configuration and Experiment Location (31, Page 1-5.19) (Sheet 2 of 2)

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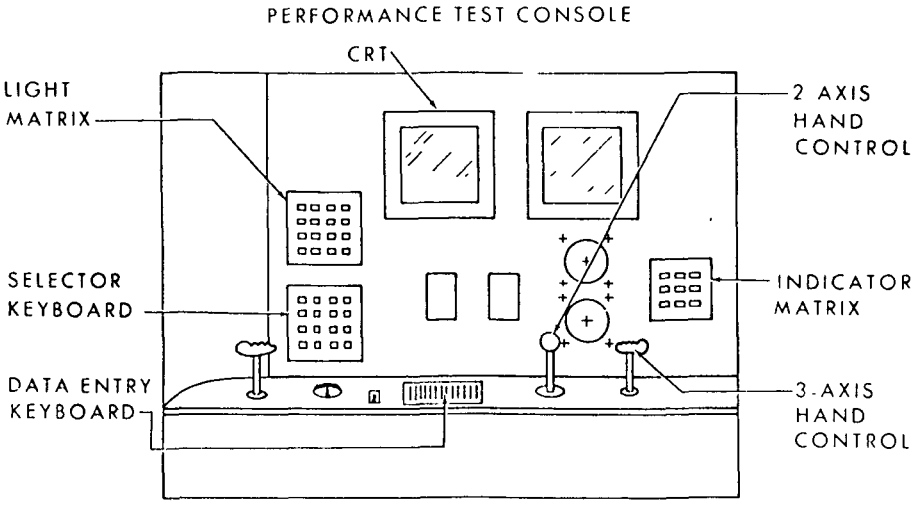
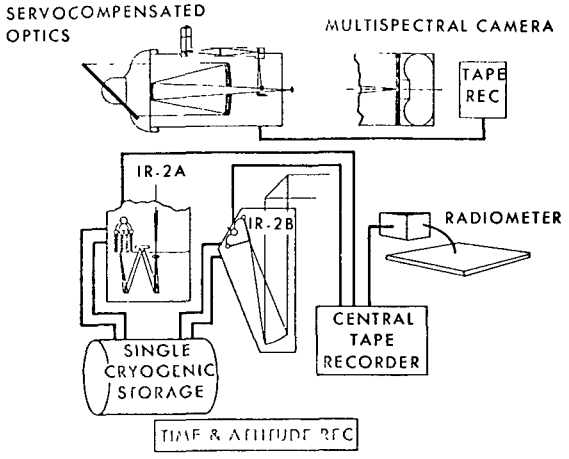
PSYCHOMOTOR FUNCTIONING - 0202



MULTI-SPECTRAL TARGET CHARACTERISTICS-0901



SYNOPTIC EARTH MAPPING - 0902

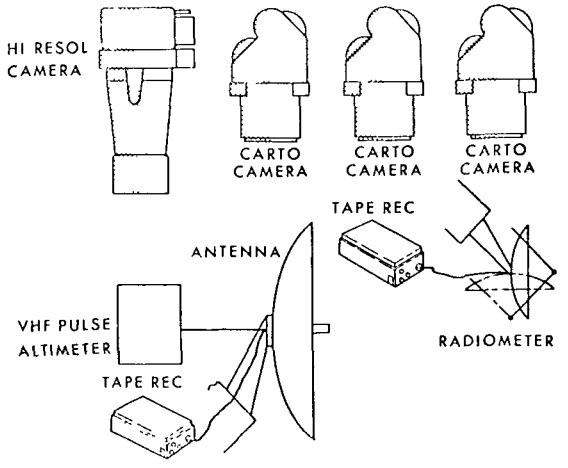


Figure 1-3. Experiment Equipment Conceptual Sketches (See 11, 13)

This flight uses an Apollo CSM fuel cell electrical generation system, the LEM ECS to provide suit circuit operation in the Lab during EVA and artificial-g operation, and selected components to provide data transmission, status, and appropriate displays.

1.2.2.3 Configuration History

The configuration history for the Intermediate Inclination Survey Mission is shown in Figure 1-4 which divides the mission profile into three major operation areas: (1) earth ascent; (2) earth parking orbit and transfer orbit; and (3) mission orbit and re-entry. The launch vehicle and spacecraft configurations are depicted for each area. Mission times from the trajectory analysis and mission phasing charts are given by relating time to the configuration changes.

1.2.2.4 Expendables Weights

1.2.2.4.1 General

The expendable weights for Mission 1 are summarized in Figure 1-5.

1.2.2.4.2 Life Support System

The following weights are ascribed to expendables of the life support system:

a. Food - The LEM has 60 pounds of food stored aboard (see 32-1-5.15), which is sufficient to sustain the crew for the mission at a consumption rate of 1.4 pounds per man per day (see 126-4,5).

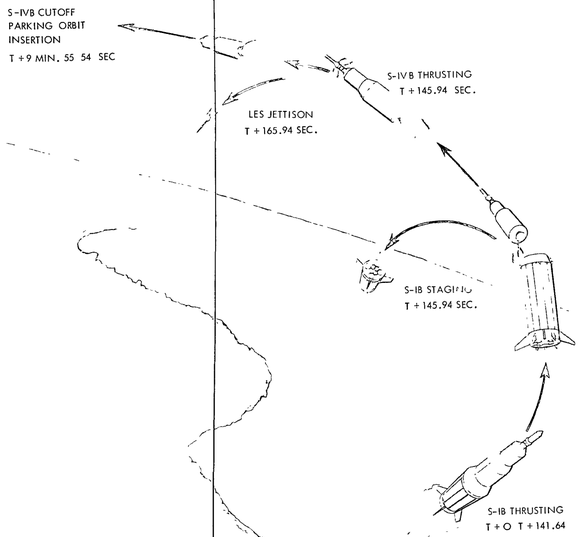
b. Oxygen - The two Block II CSM oxygen tanks are filled to 324.7 pounds (see 31-1.5-10).

c. Hydrogen - The two Block II CSM hydrogen tanks are filled to 58.24 pounds (see 80-167). The LEM two Block II LEM hydrogen tanks are filled to a total capacity of 41.3 pounds (see 31-1-5.10).

d. Repressurization - For extravehicular activity, the CM atmosphere is evacuated overboard. It was assumed that the CM and LEM have no airlocks and that the LEM atmosphere is also dumped. LEM repressurization requires 6 pounds of oxygen (see 127); the CM requires 7 pounds (see NAA Training Manual). Five repressurizations are required during the mission (see 31-1-5.10).

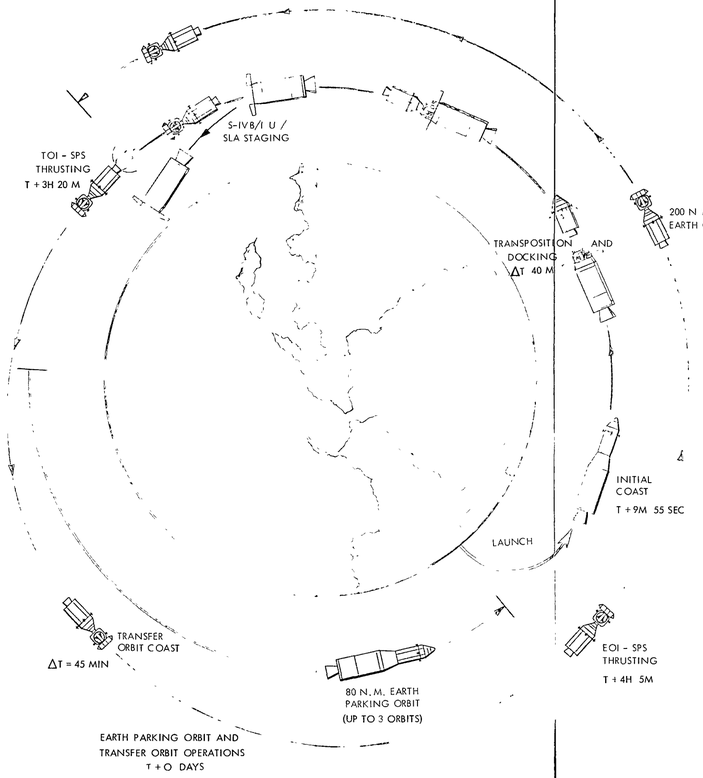
e. Leakage - The oxygen leakage rate for the CM is 5.66 pounds per day (see 126); for the LEM it is 4.8 pounds per day. The total oxygen

MISSION 1 INTERMEDIATE INCLINATION SURVEY



1-27-1

1-27-2



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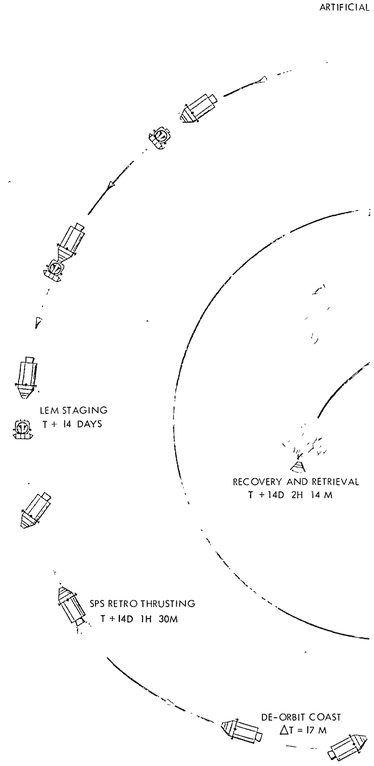

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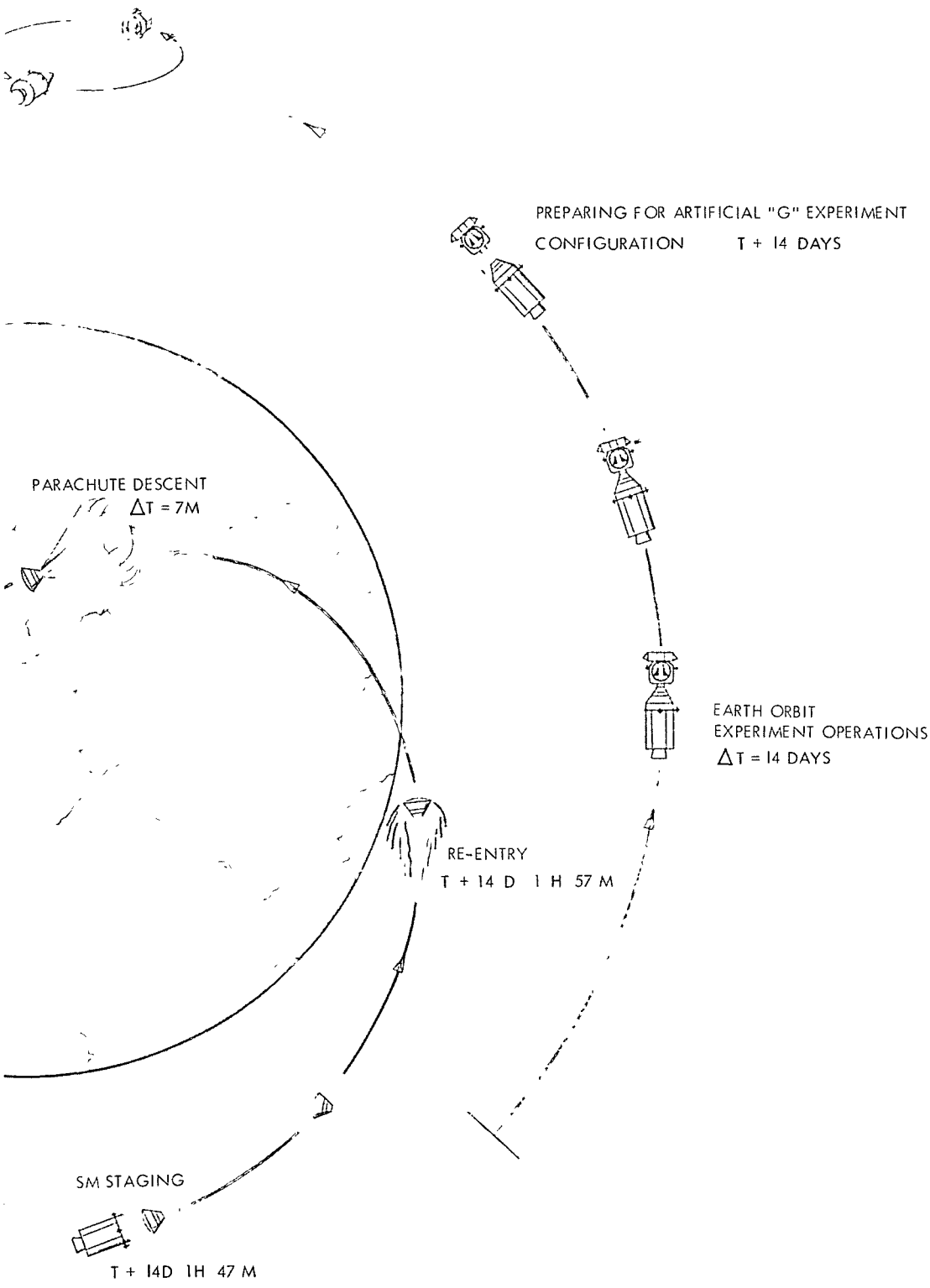

MISSION ORBIT AND RE-ENTRY OPERATIONS

T + O DAYS TO T + 14 DAYS

Figure 1-4. Configuration History for the Intermediate Inclination Survey

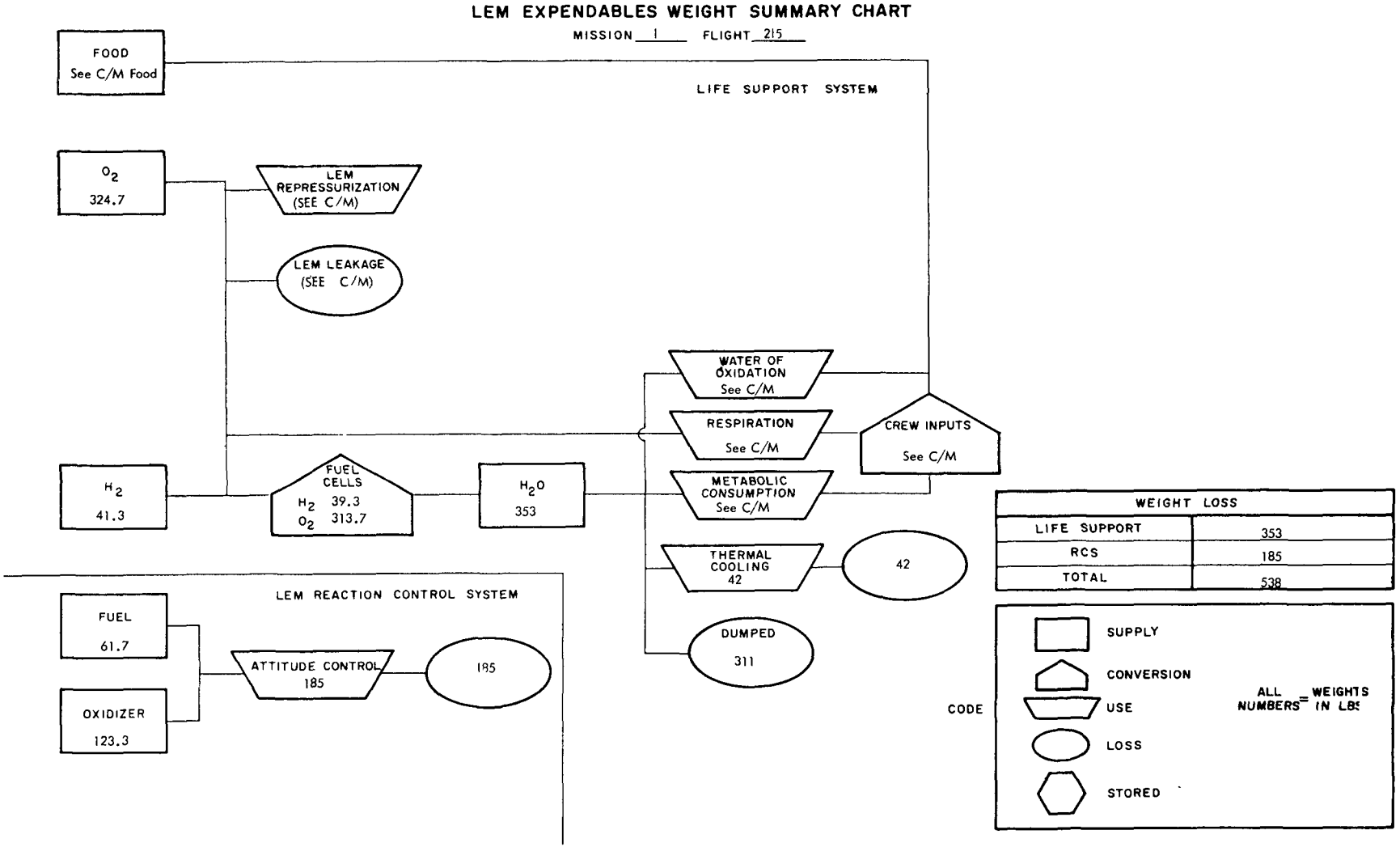

Figure 1-5. Expendables Weight Summary Chart (Sheet 1 of 2)

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MISSION 1 FLIGHT 215

CSM EXPANDABLES WEIGHT SUMMARY CHART

WEIGHT LOSS	
LIFE SUPPORT	577.5
SM RCS	835
CM RCS	268.8
SPS	2200
TOTAL	3881

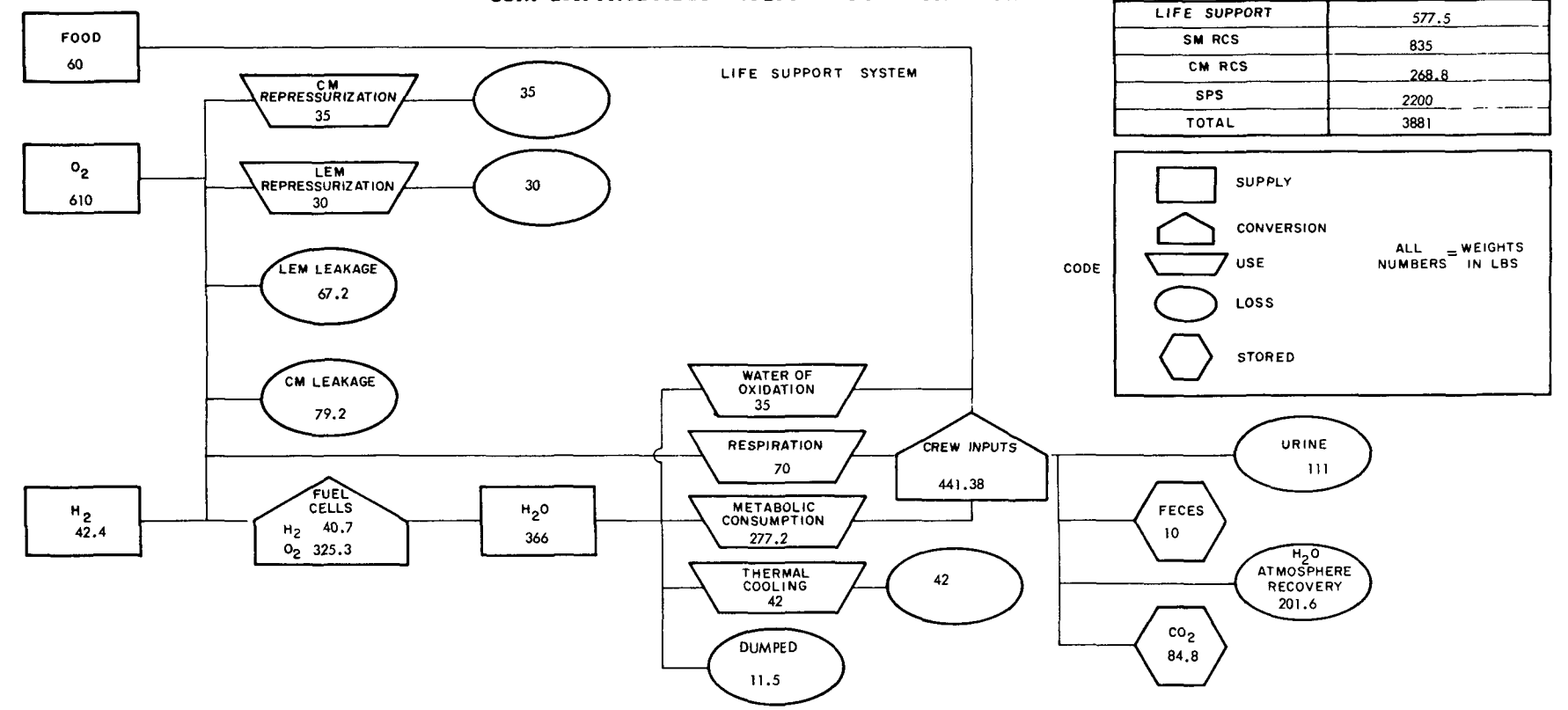

Figure 1-5. Expendables Weight Summary Chart (Sheet 2 of 2)

leakage is 67.2 pounds for the LEM and 79.2 pounds for the CM, based on a cabin pressure of 5 psia.

f. Respiration - The oxygen required for crew respiration is 1.67 pounds per man per day (see 126), which results in a requirement of 43 pounds for the CM and 27.1 pounds for the LEM, or a total of 70.1 pounds.

g. Fuel Cells - The CSM has three fuel cell assemblies (see 80); the LEM has two (see 31-1-5.10). The hydrogen and oxygen supply for the CSM cells is the original supply less amounts required for respiration, repressurization, and leakage replacement.

h. Water - The fuel cell production water varies from 28.8 to 82.8 pounds per day with power requirements for each of the three assemblies. The total weight of hydrogen and oxygen released to the fuel cells combines to form water.

i. Water of Oxidation - The water needed to reconstitute powdered food is 0.035 pounds per man per hour (see 126), or a total of 35.38 pounds.

j. Metabolic Consumption - Drinking water consumed is 0.275 pounds per man per hour (see 126), or a total of 277.2 pounds.

k. Thermal Cooling - Water required for LEM cooling is 42 pounds (see 31-1-5.10). An equal amount was assumed for the CM.

l. Water Dumped - Excess water produced by the fuel cells is vented overboard.

m. Crew Inputs - The total weight inputs from the crew are as follows:

(1) The urine produced is 0.11 pounds per man per hour (see 126), or a total amount of 111 pounds, which is vented overboard.

(2) Feces, which are produced at a rate of 0.24 pounds per man per day (see 126), total 10.1 pounds and are stored aboard.

(3) Water vapor, from expiration and perspiration, is emitted at a rate of 0.2 pounds per man per hour (see 126). The total of 201.6 pounds is recovered and dumped.

(4) Carbon dioxide from human respiration is produced at a rate of 2.02 pounds per man per day (see SID 63-1370-2) and totals 84.8 pounds; the carbon dioxide is absorbed by lithium hydroxide filters.

1.2.2.4.3 SM-RCS

The four Block II SM-RCS fuel tanks (see 80-viii) are loaded to 282.2 pounds. The four Block II SM RCS oxidizer tanks (see 80-viii) are loaded to 554.8 pounds.

The RCS requirement for attitude control during the experiments is 614 pounds of fuel-oxidized mixture in a ratio of one to two. The remaining reactants are available for normal attitude control functions, for CSM booster separation, for ullage maneuvers prior to SPS ignition, and for separation of the SM and the CM.

1.2.2.4.4 CM-RCS

The two Block II CM-RCS fuel tanks (see 80-85) are filled to 90.4 pounds. The two Block II CM-RCS oxidizer tanks are filled to 178.4 pounds.

There are 225 pounds of reactants required to provide the following:

a. Maneuver capabilities during entry after SM separation.
b. Maneuver capabilities in the event of a high-altitude abort after LES jettison.
c. Three-axis control to stabilize CM motion prior to deployment of LES parachutes.

Fuel and oxidizer left in the tanks after parachute deployment and final attitude correction is dumped before touchdown.

1.2.2.4.5 SPS

The one Block II SPS fuel tank (see 80-viii) has a capacity of 15,435 pounds; the fuel load for this flight is 797 pounds. The one Block II SPS oxidizer tank has a 31,620-pound capacity; the load for this flight is 1,594 pounds.

A reactant weight of 747 pounds is used for transfer orbit injection, to transfer from the 80-n. mi. orbit to the 200-n. mi. orbit; 730 pounds is used for the 200-n mi. mission orbit injection (see 111). A reactant weight of 723 pounds is used by the SPS for the retro maneuver (calculated).

1.2.2.4.6 LEM RCS

The two Block II LEM RCS fuel tanks (see 31-1-5.10) have a 204.8-pound total capacity; the load for this flight is 62 pounds. The two Block II LEM RCS oxidizer tanks hold 552.4 pounds (see 31-1-5.10); the load for this flight is 124 pounds.

Of the total, 185 pounds of reactant is required for LEM attitude control (see 31-1-5.15).

1.2.2.5 Weight Profile

The vehicle weight changes for the mission from S-IVB burnout through CM re-entry are shown in Figure 1-6. Data for this chart was taken from reference 111 of the Weight Summary Chart (Table 1-1) and from the Expendables Weight Summary Charts (Figure 1-5).

Mission 1 vehicle weight at S-IVB burnout is 67,391 pounds. S-IVB staging and jettisoning of the LEM adapter reduces the weight to 32,406 pounds. Fuel expenditure for transfer orbit injection and mission orbit injection brings the weight down to 30,929.

The payload weight required for this mission is 30,215 pounds. The differential between maximum vehicle weight and payload weight is shown as "margin," which for this mission is 714 pounds.

During the 14-day earth orbit coast, the consumption of expendables such as food, water, oxygen, hydrogen, and RCS reactants reduces the spacecraft weight to 28,265 pounds.

LEM staging brings the CSM weight to 19,723 pounds at the beginning of retrofire. Fuel consumption for retro and SM staging then reduces the weight to 10,465 pounds at CM re-entry.

1.2.3 CREW ANALYSIS

1.2.3.1 General

Crew tasks during Mission 1 are divided into two basic categories: tasks concerned with flight operations, and those concerned with the conduct of the experiments. The crew flight operations required from liftoff to the 200-n. mi. orbit and from initiation of deorbit to touchdown are detailed in Appendix D. Flight operations activities required of the crew during the 200-n. mi. orbit consist primarily of system status

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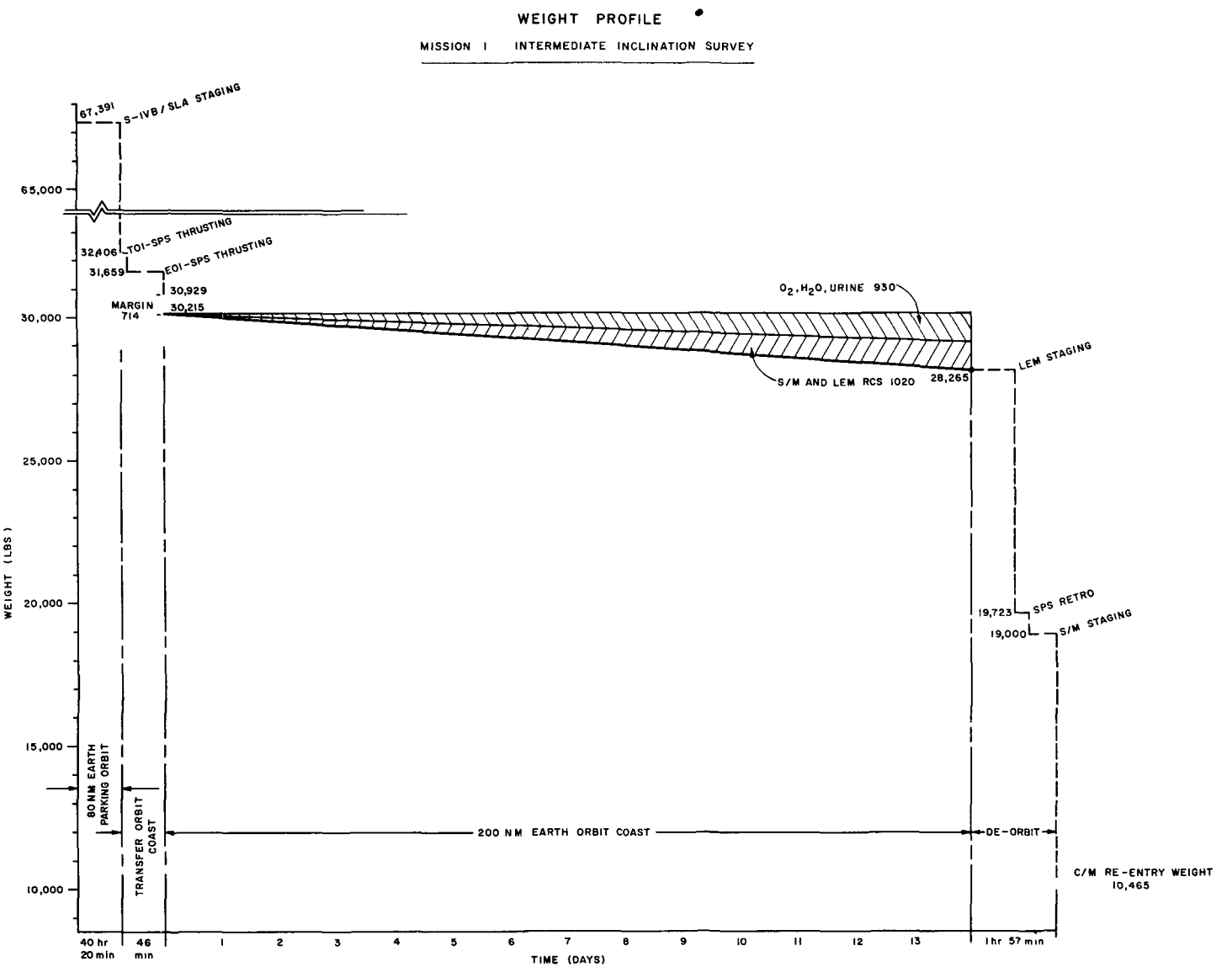


Figure 1-6. Weight Profile Mission 1

checks, IMU alignments, and ground communications. Table 1-5 shows a summary of the system status check tasks as adapted from the Apollo Design Reference Mission (see 25). The IMU alignment task times was taken directly from this source. A detailed task analysis of the crew activities required for conduct of the experiments is presented also in Appendix D.

Analysis of the flight crew operations, and the sleep, eating, hygiene, and rest requirements for a typical orbital day indicated that approximately 22.5 man-hours are available in a 24-hour day to perform experiments. This allows a total of approximately 315 man-hours available for experiments over the 14-day mission. Utilizing the number of experiment sequences and the number of test subjects (see 31, 32, 33) recommended for the assigned experiments with the experiment times as outlined in Appendix D, it was determined that approximately 572 man-hours would be required to conduct the assigned experiments. Therefore, an experiment priority system was established. The priority classifications scheme and priority numbers are as follows:

a. First priority experiments are those that satisfy primary mission objectives.
b. Second priority experiments are those that contribute to the assessment of physical health or well being of the astronauts.
c. Third priority experiments are those that satisfy secondary mission objectives.
d. Fourth priority experiments are the biomedical and behavioral experiments that contribute general scientific knowledge.
e. The lowest priority experiments that have been performed on previous flights and are of such a nature that omission is not detrimental to the mission objectives.

It should be noted that the priority classification scheme and assignments were arbitrarily made due to the excessive experiment time load. They do not constitute a recommended or suggested solution to the experiment priority and integration problem.

The flight operations and experiment scheduling assumptions utilized in scheduling Flight 215 are outlined in paragraphs 1.2.3.2 and 1.2.3.3. No communication times for flight operations were programmed on this flight because the communications contact points were not known at the time of scheduling.

Table 1-5

Earth Orbital System Status Checks

Task	Frequency of Performance	Duration (Min)
CM		
Check ECS	Hourly	1
Check EPS	Hourly	3
Check SM-RCS	Every 4 hours	4
Check CM-RCS	Every 12 hours	4
Replace CO_2 and odor absorber	Every 12 hours	8
Charge entry and post-landing batteries	Every 6 hours	3
Purge fuel cells	Every 8 hours	3
Check SPS	Every 4 hours	4
LEM		
Check ECS	Hourly	1
Check PGS	Hourly	1
Check PDS	Hourly	1
Replace CO_2 and odor absorber	Every 12 hours	8
Check LEM-RCS	Every 12 hours	4
4.75 man-hours required for system status checks in a 24-hour day.		

CM tasks were adapted from translunar and transearth coast phase (see Appendix C and 24).

LEM tasks were adapted from lunar surface phase (see Appendix C and 24).

The actual schedules are presented in Figure 1-7. Table 1-6 presents the total amount of time scheduled for each of the experiments, the number of experiment sequences actually scheduled, and the number recommended (see 31, 32, 33) for the 14-day flight.

The following time utilization chart shows the percentage of time actually scheduled for the 4 events indicated and presents the total man-hours of experiments scheduled. This shows that approximately 54 percent of the recommended experiment time was actually scheduled for the flight.

MISSION 1 TIME UTILIZATION

Experiments	31.0 percent
Flight Operations	13.7 percent
Personal Time	54.3 percent
Unscheduled Time	1.0 percent
Total Experiment Time Recommended*	572 man-hours
Total Experiment Time Scheduled	310.7 man-hours

1.2.3.2 Flight Operations Scheduling Assumptions

The astronaut day will consist of 10 hours on duty, and 14 off duty. Of the 14 hours off duty, six hours will be devoted to sleep, five hours to rest, and three hours to eat and for personal hygiene.

Eight IMU alignments are required each 24 hours. Tht total includes five additional alignments required by experiments 901 and 902.

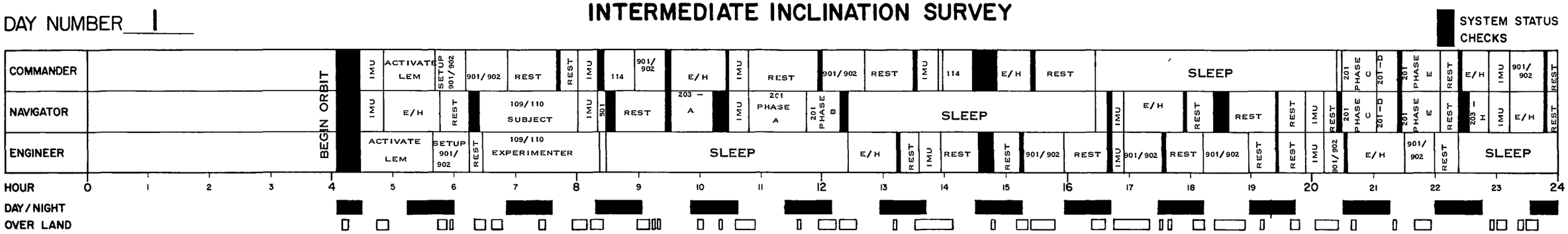
System status checks for flight operations require 4.75 man-hours during a 24-hour period (AMPTF adapted). Any system checks required for experiments are included in the experiment setup times.

*Grumman number of experiment sequences and subjects tested (see 38, 39, 40). Experiment times from Appendix D.

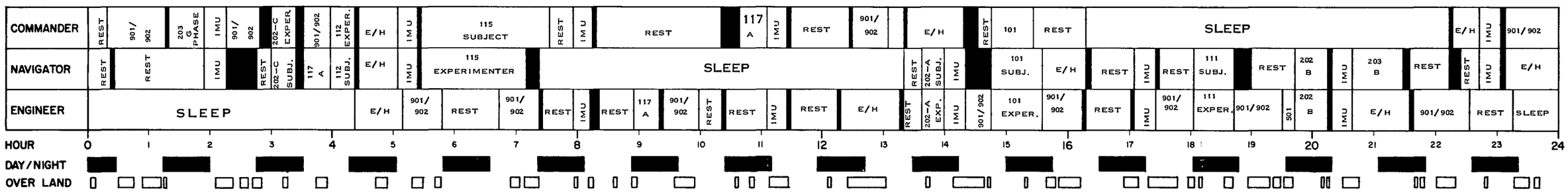
MISSION NUMBER 1

INTERMEDIATE INCLINATION SURVEY

DAY NUMBER 1



DAY NUMBER 2



DAY NUMBER 3

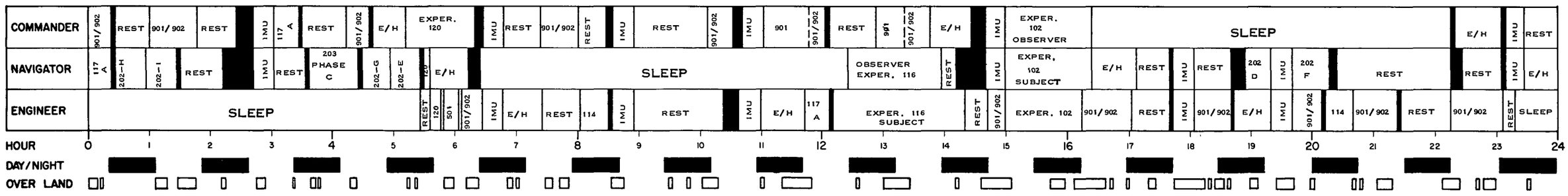
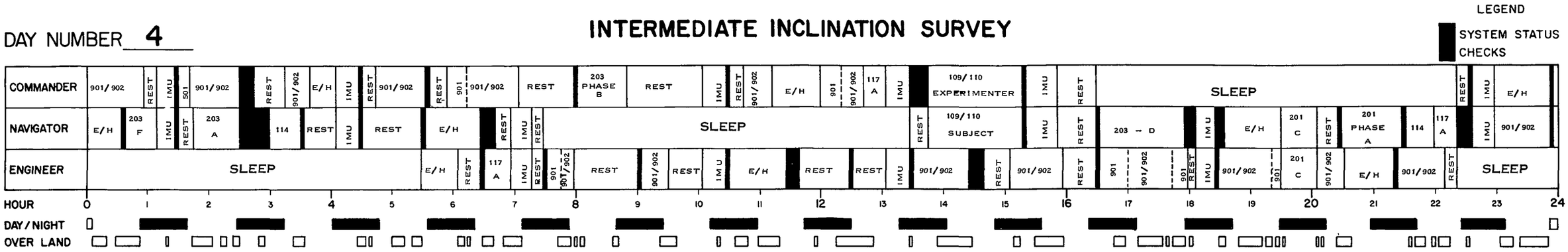


Figure 1-7. Crew Scheduling
(Sheet 1 of 5)

MISSION NUMBER 1

DAY NUMBER 4

INTERMEDIATE INCLINATION SURVEY



DAY NUMBER 5

DAY NUMBER 6

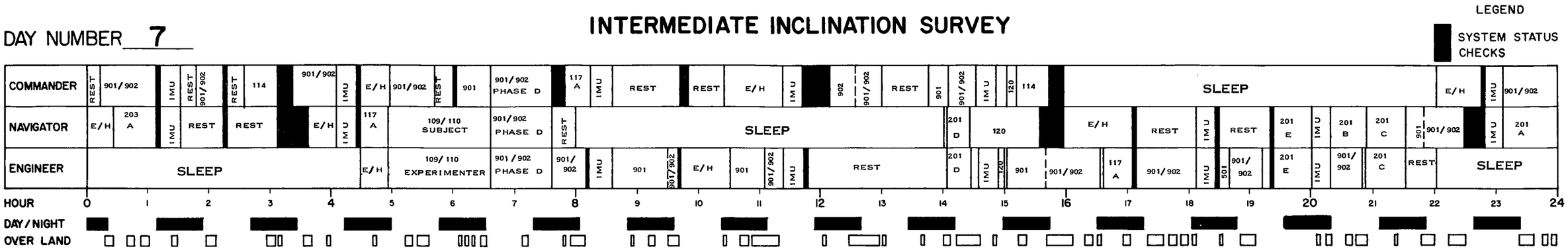
Figure 1-7. Crew Scheduling
(Sheet 2 of 5)

1-40-2

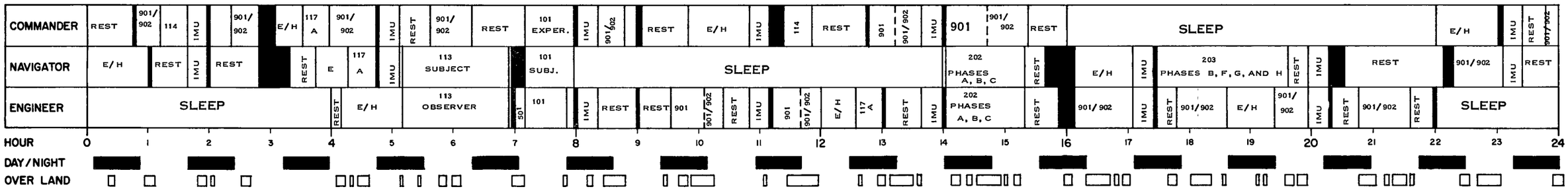
MISSION NUMBER 1

DAY NUMBER 7

INTERMEDIATE INCLINATION SURVEY



DAY NUMBER 8



DAY NUMBER 9

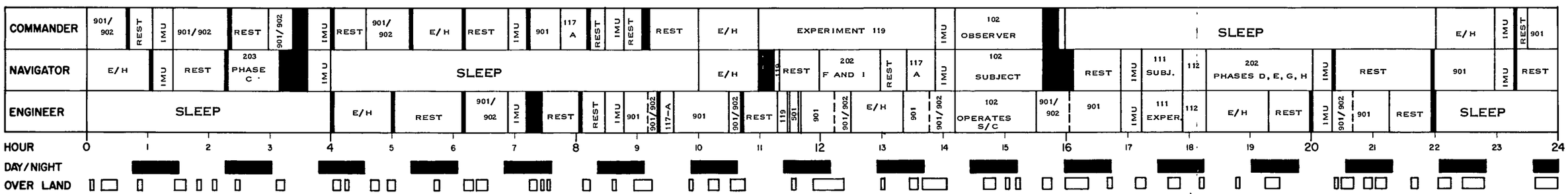


Figure 1-7. Crew Scheduling
(Sheet 3 of 5)

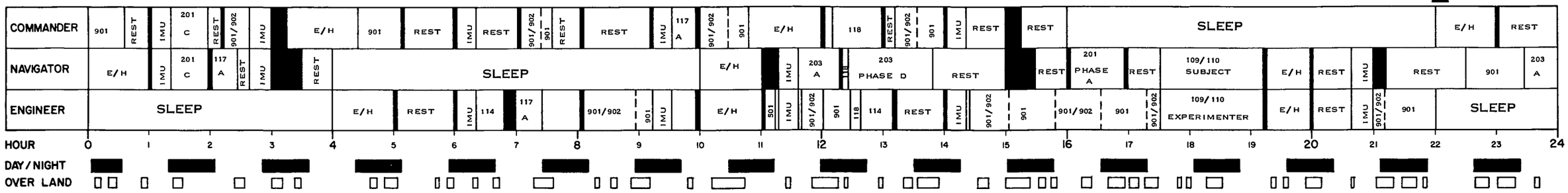
MISSION NUMBER 1

INTERMEDIATE INCLINATION SURVEY

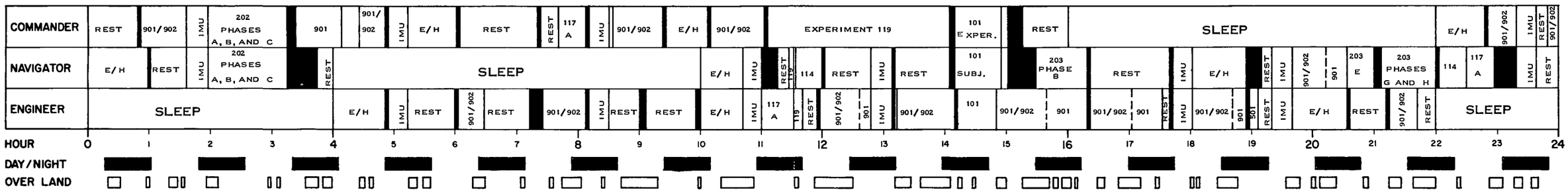
LEGEND

SYSTEM STATUS CHECKS

DAY NUMBER 10



DAY NUMBER 11



DAY NUMBER 12

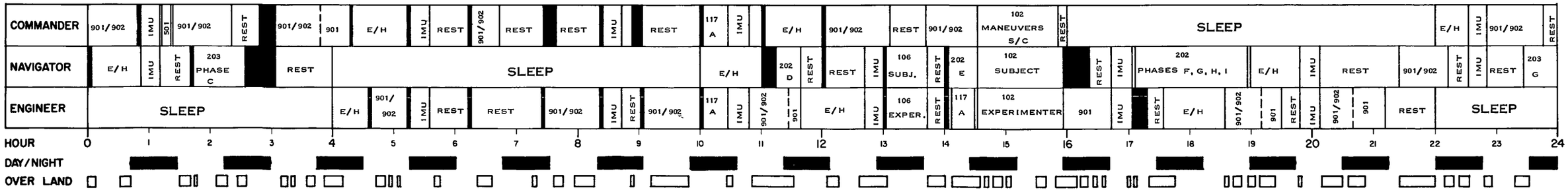


Figure 1-7. Crew Scheduling
(Sheet 4 of 5)

1-42-2

MISSION NUMBER 1

INTERMEDIATE INCLINATION SURVEY

LEGEND
SYSTEM STATUS CHECKS

DAY NUMBER 13

| | | |

Table 1-6

Summary of Scheduled Experiment Times and Sequences

Experiment Number	Man-Minutes Expended	Total Number of Sequences	
		Actual (14 Days)	Recommended (14 Days)
0101	532	4	4
0102	992	4	4
0103	-	0	2
0104	-	0	2
0105	-	0	2
0106	78	1	10
0107	-	0	11
0108	-	0	2
0109			
0110	1010	5	4
0111	-	0	2
0112	92	2	2
0113	214	1	1
0114	324	12	12
0115	666	3	6
0116	217	1	1
0117A	800	12	14
0117B	-	0	2
0118	63	1	1
0119	380	2	2
0120	166	2	3
0121	-	0	1
0201A	270	5	4
0201B	96	3	2
0201C	380	5	4
0201D	126	3	2
0201E	228	3	2
0202A	126	4	4
0202B	160	5	4
0202C	184	4	4
0202D	96	4	4

Table 1-6

Summary of Scheduled Experiment Times and Sequences (Cont.)

Experiment Number	Man-Minutes Expended	Total Number of Sequences	
		Actual (14 Days)	Recommended (14 Days)
0202E	112	4	4
0202F	128	4	4
0202G	96	4	4
0202H	112	4	4
0202I	112	4	4
0203A	205	5	4
0203B	192	4	4
0203C	196	4	4
0203D	172	2	2
0203E	54	2	2
0203F	112	4	4
0203G	124	4	4
0203H	84	4	4
0301	440	1	1
0501	140	14	28
0901	2416	Continuous*	Continuous*
0901/0902	6823	Continuous*	Continuous*

*See Experiment Assumption Number 1.

References 31, 32, 33, 24, 25.

1.2.3.3 Experiment Scheduling Assumptions

Experiment 0901 and 0902 are scheduled for periods of daylight in which the spacecraft is over a land mass for at least five minutes. During those phases, the experiment is not activated until land is sighted. The experiment remains active (crew monitoring continues as long as land crossings continue). For several selected orbits, experiment 0902 is scheduled over land in darkness.

Experiment 301 has first priority on the 14th day of the mission.

1.2.3.4 Experiment Priority Assumptions

The assumed experiment priorities used in preparing the crew schedules are as follows:

Priority	Experiment Number and Name	
1	0901	Multispectral Target Characteristics
1	0902	Synoptic Earth Mapping
2	0501	Radiation Environment Mapping
2	0100 Series	Biomedical - 0109, 0110, 0111, 0112, 0113, 0114, 0118, 0101, 0102, 0115, 0116, 0119, 0120, 0117A, 0106.
3	0301	Artificial-g
4	0200 Series	Behavioral - 0201, 0202, 0203

1.2.4 DATA FLOW

1.2.4.1 General

Data flow characteristics - data recording and transmission, magnetic tape storage, and data collection - are detailed in the following paragraphs. The data flow was profiled for Mission 1 using the analysis presented in Part II, Section 3, of this report. Instructions for using the profiles are included in the section under the applicable heading.

1.2.4.2 Data Recording and Transmission

A data recording and transmission profile for the first three days of Mission 1 is presented in Figure 1-8.

The top line shows the distribution of experiment data collection sequences and their duration in time. The second line shows the distribution of MSFN S-band telemetry ground station coverages and their durations. The third line shows the coverage distribution of those telemetry stations assumed to have television receiving capability (Goldstone, Madrid, and Canberra). The fourth line shows the distribution of station contacts tentatively assigned for the reception of dumped operational data (and low-sampling-rate experimental data) from the CSM recorder.

Note

These contacts were assigned on a one-dump-per-orbit basis with no interval greater than 120 minutes whenever possible. When intervals greater than 120 minutes occurred, an extra operational contact was tentatively assigned to allow the dumping of the extra reel of stored operational data tape. Whenever a choice was available, the operational contact was made over the U.S.

The fifth line is a profile of the rate of accumulation of the magnetic tape being used to record high-rate PCM data in the LEM, compared to the rate at which reels of this tape may be dumped to the ground telemetry stations. Each upward-pointing arrow indicates the accumulation of one full reel of tape (30 minutes recording time) and each downward-pointing arrow indicates the completed dump of one full reel of tape (30 minutes dumping time).

Note that the profile for Mission 1 shows a great amount of recording activity during this period, much more than can be dumped with the presently planned spacecraft data transmission capabilities. This excess is readily apparent by observing the number of upward-pointing arrows, indicating the number of reels recorded, versus the number of downward-pointing arrows, indicating the number of reels dumped in any one day.

It should also be noted that a non-coverage interval greater than 120 minutes will occur on the average of once per day with the presently planned network of 14 S-band telemetry ground stations.

1.2.4.3 Magnetic Tape Storage Profile (See Figure 1-9)

For each applicable mission a sample magnetic tape storage profile was developed for the first three days of that mission to show the nature of the net accumulation of reels of recorded magnetic tape. This profile is derived directly from the fifth line on the

data recording and transmission profile. Figure 1-9 is the tape storage profile for Mission 1, from which it is readily apparent that a daily net accumulation of reels of recorded tape exists and that the rate of accumulation tends to be cyclical.

1.2.4.4 Data Collection Summary

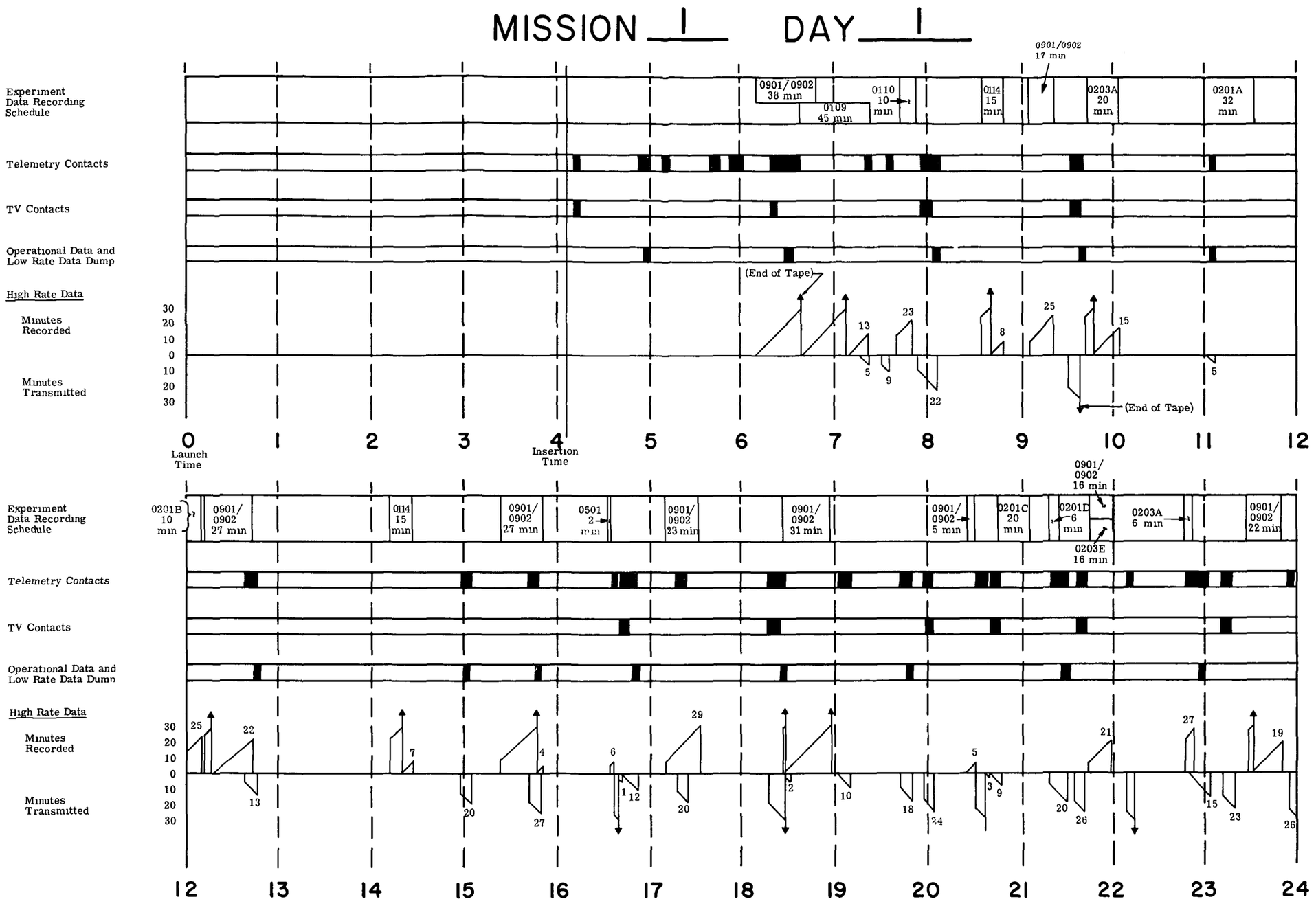
The data collection requirements for Mission 1 are summarized in Table 1-7.

The first column designates each experiment by experiment number (see 38, 39, 40, 77). The exception to this is series 0100, the biomedical experiments, and series 0200, behavioral experiments. The next column is the maximum number of special analog channels which must be recorded at any one time because of that experiment. The next indicates whether or not the digital output of a PCM multiplexer must be recorded.

The third column indicates the minutes of recording time required by each experiment sequence. The fourth and fifth columns indicate the total number of sequences and total minutes of recording time respectively required by that experiment during the course of the mission.

The next column contains a graphically-derived value of the total transmission time available during the mission and is entered only in the over-all totals. The figure given for "Recorded Backlog" is the difference between the over-all total "high rate" PCM and analog data recording time, and the over-all total available transmission time. Further experiment data collection details are given in Part II, Section 3.

Several forms of data are required on Mission 1 for which no detailed estimates of weight and volume were available. For experiments in which estimates were available, the chart shows that both the film and the excess magnetic tape requirements exceed the assumed recoverable payload weight of 250 pounds. This is discussed in Part II, Section 3.



DATA RECORDING AND TRANSMISSION PROFILE

Figure 1-8. Data Recording and Transmission Profile - Mission 1 - Day 1 (Sheet 1 of 3)

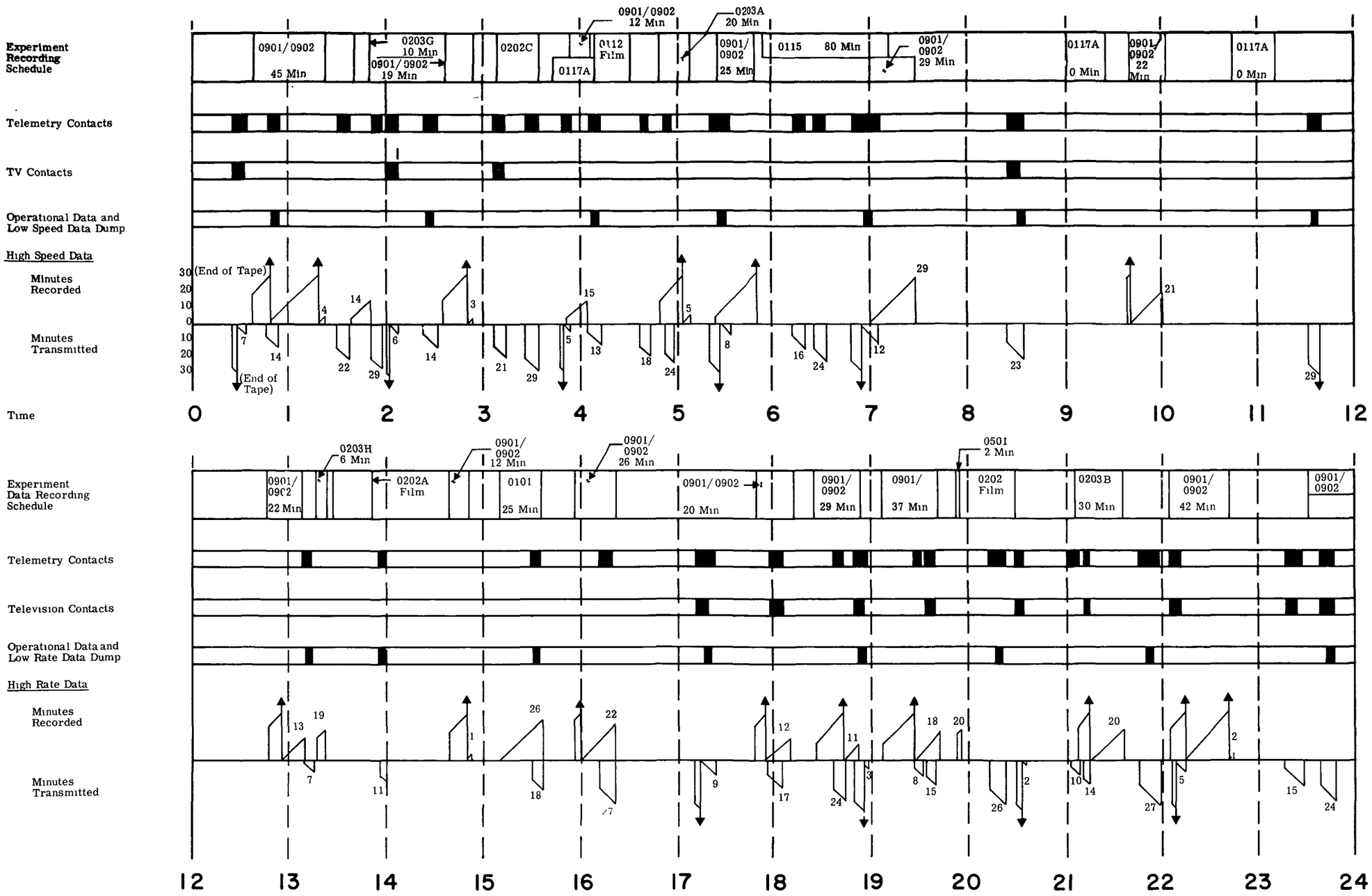
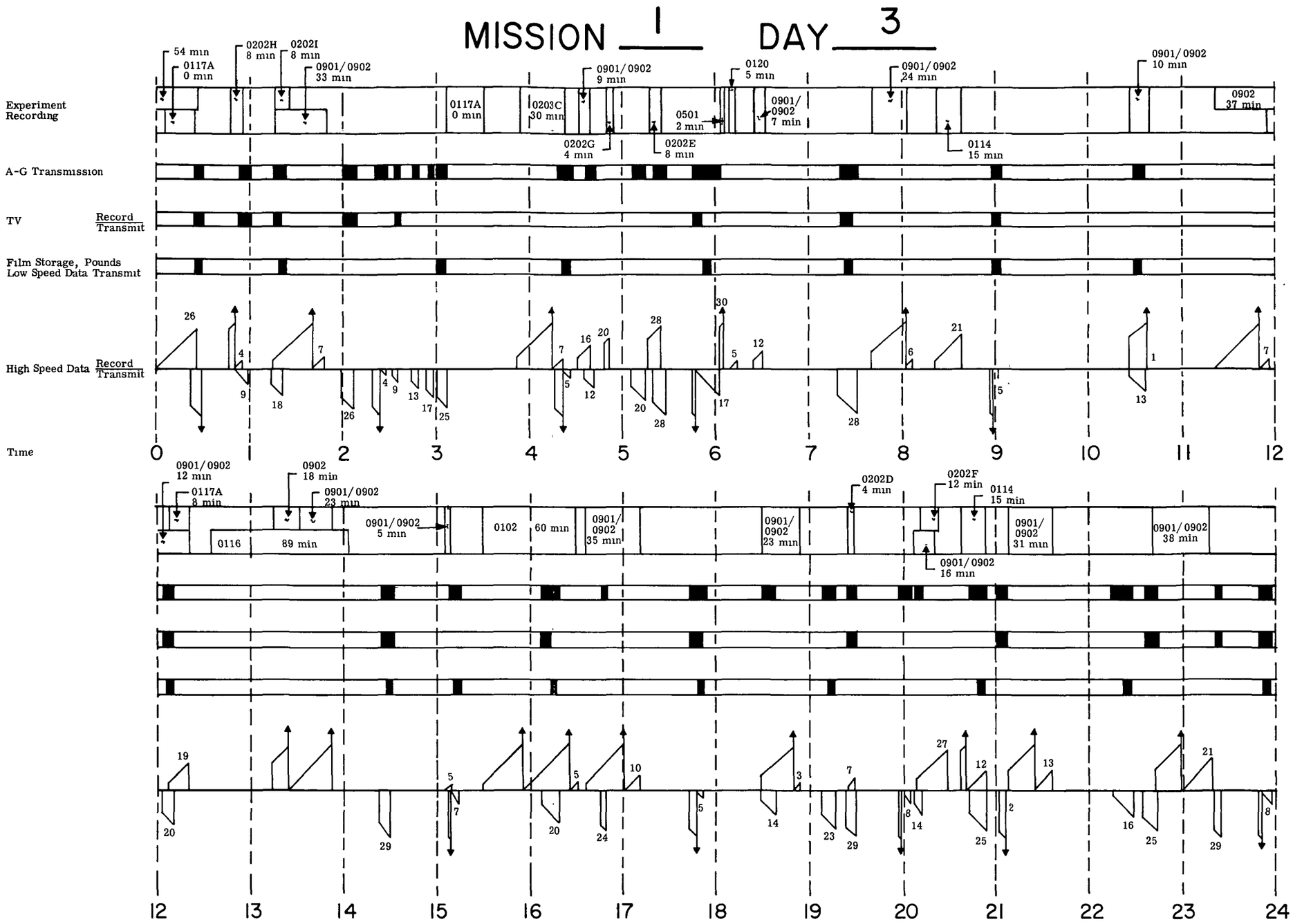


Figure 1-8. Data Recording and Transmission Profile - Mission 1 - Day 2 (Sheet 2 of 3)

DATA RECORDING AND TRANSMISSION PROFILE



DATA RECORDING AND TRANSMISSION PROFILE

Figure 1-8. Data Recording and Transmission Profile - Mission 1 - Day 3 (Sheet 3 of 3)

MAGNETIC TAPE STORAGE PROFILE
MISSION I

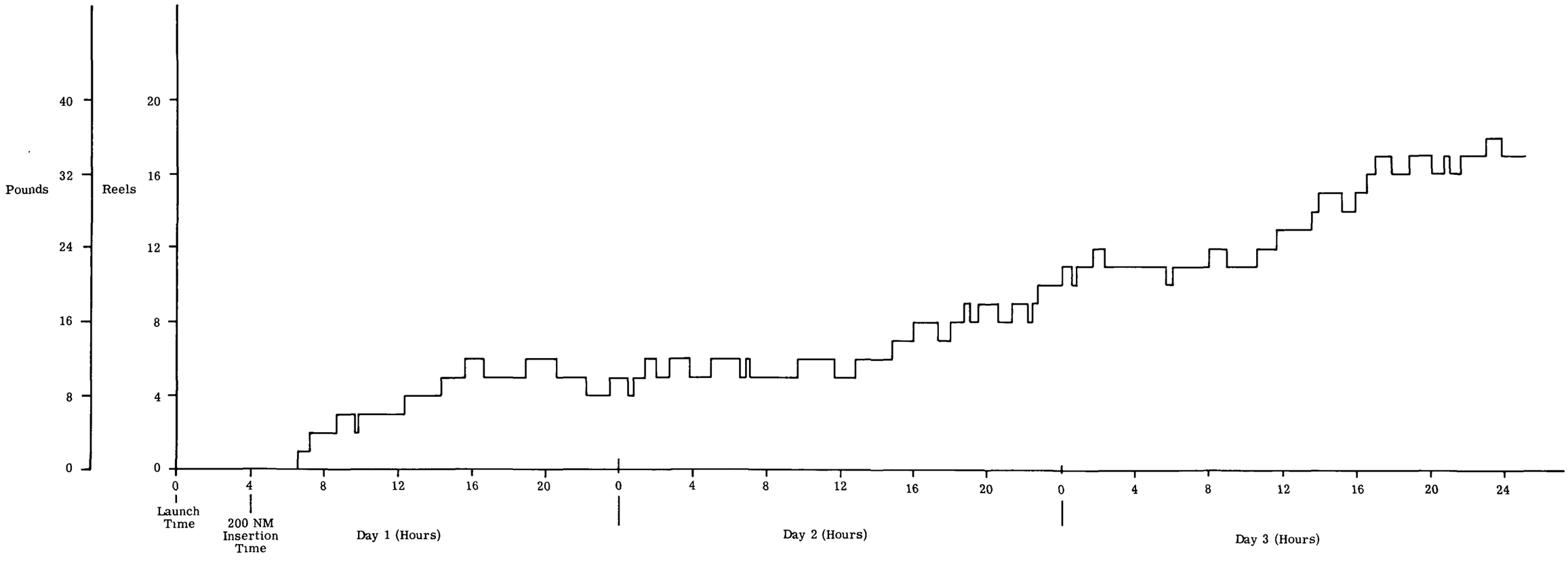


Figure 1-9. Magnetic Tape Storage Profile - Mission 1

Table 1-7

Mission 1

Data Collection Summary

Experiment	Data Recording							Recovery Requirements				Remarks
	Max. No. of Channels in Use at One Time		Minutes per Sequence	Total No. of Sequences	Total Minutes Recorded	Transmission Time Avail. (Minutes)	Recorded Back Log (Minutes)	Magnetic Tape		Pounds of Film	Miscellaneous	
	Anal.	Dig.						Reels	Lbs.			
	*	*				**	***	**	**			* Assumed time sharing of channels as required. ** Summed only for total mission. ***Difference - Recorded minus Transmitted - only for total mission.
0100	6	1	*	29**	1207					X	X	* Varies for each experiment in series (see Data Collection Charts, Part II, Section 3. **Varies for each experiment in series; number (29) is total for series for which recorder is used.
0200	1	1	*	51**	790					48+	X	* Varies for each experiment in series (see Data Collection Charts, Part II, Section 3. **Varies for each experiment in series; number (51) is total for series for which recorder is used.
												X indicates a requirement exists but no information is available.

Mission 1

Table 1-7

Data Collection Summary (Cont.)

Experiment	Data Recording							Recovery Requirements				Remarks
	Max. No. of Channels in Use at One Time		Minutes per Sequence	Total No. of Sequences	Total Minutes Recorded	Transmission Time Avail. (Minutes)	Recorded Back Log (Minutes)	Magnetic Tape		Pounds of Film	Miscellaneous	
	Anal.	Dig.						Reels	Lbs.			
0501		1	2	14	28						X	
0301	1	1	150*	1	150					X		*Estimated.
0901 0902	6	2	24*	184	4384					300+		*Average minutes per sequence. +Estimated by taking 1/3 of total 0901/0902 film requirements for a 45-day mission.
0901	6	1	23*	67	1545					X**		* Average minutes per sequence. **Included in the 300 pounds for 0901/0902
Total	9*	2*			8104	3919	4185	140	280	300+	X	*Total number of channels in use at any one time during mission, not a column summation; assumes time sharing as required.

1.2.5 TRAJECTORY

1.2.5.1 Description

The trajectory consists of a launch from AMR at an azimuth of 44 degrees to an initial parking orbit at 80-n. mi. altitude (see 49), inclined at 50 degrees to the equatorial plane (see 19). In choosing a parking orbit, the lowest possible altitude provides the greatest payload capability (see 84-2). The 80-n. mi. orbit (launch trajectory to 80 n. mi. was obtained from Reference 49) represents a satisfactory compromise for a reasonable orbit life time and utilizes all of the allocated S-IVB fuel without the necessity for restart.

The parking orbit insertion point for the ground traces was taken from Reference 49. Insertion characteristics are the following:

Latitude 39.37 degrees	Longitude 65.93 degrees West
Relative Azimuth 53.4 degrees	Velocity 25,665 feet per second
Time 12:00 Hours + 595.54 Seconds	Altitude 80 n. mi.

A Hohmann transfer from the parking orbit to the final mission orbit altitude of 200 n. mi. was used for propulsion economy (see 19). The 200-n. mi. altitude was selected to guarantee an orbital lifetime in excess of 45 days, using pessimistic atmosphere models (see 17).

A Hohmann transfer to the mission altitude of 200 n. mi. is affected by the SPS (see 19). The parking orbit time was chosen after considering minimum crew activity in orbit and requirements for transfer maneuver tracking (Reference 91 requires tracking within seven minutes of burn).

The re-entry trajectory was obtained from Reference 49. This data is based on a re-entry vacuum perigee of 0-n. mi. altitude.

1.2.5.2 Assumptions

The trajectory characteristics for Mission 1 were based on the following assumption: A launch at 12 GMT was selected to determine the nominal mission trajectory.

1.2.5.3 Characteristics

Table 1-8 is a listing of the flight dynamics parameters during powered flight. Items 1-9 and 14-16 are taken from Reference 49, Table 10. These parameters are the basic control times and weights for the mission. Item 13, vehicle weight is the payload delivered to the mission orbit.

Figure 1-10 is a plot of axial acceleration, relative flight path angle, relative velocity, and altitude as a function of time during launch.

Figure 1-11 is a plot on a uniform grid of the launch and re-entry phase ground trace for Mission 1. Abort areas paralleling the Gemini abort areas are shown. Figure 1-12 plots illumination and radar coverage during the launch phase. For the ground trace of the later orbits, the uniform grid with ascending node location indicated can be used with a transparent overlay to generate the ground trace of any desired orbit. This material is included in Volume 4.

1.2.6 ABORT ANALYSIS

1.2.6.1 General

Four major abort modes are applicable for Mission 1 (see 24, 25, 26, 105, 106):

LES Abort
Suborbital Abort
Abort to Orbit and
De-orbit Abort.

Ground rules for each of the major abort modes are defined in the following paragraphs, and abort mode descriptions and abort logic diagrams are presented for each mode. The relationship between the mission phases and the abort mode available at any particular point within the phase is shown in Figure 1-13.

1.2.6.2 Launch Escape System Abort

1.2.6.2.1 General

A Launch Escape System (LES) abort may be performed any time between LES enablement during prelaunch and LES jettison during the second stage thrusting period. The following paragraphs delineate the ground rules and mode descriptions of LES aborts.

Table 1-8

Saturn IB Reference Trajectories; 200-N. Mi. Circular Orbit by Hohmann Transfer
(Perigee = 80 N. Mi., i = 50.3)

Trajectory Event	Time (seconds)	Geodetic Latitude (degrees)	Longitude (degrees)	Velocity [1] (feet per second)	Altitude (feet)	Flight Path Angle [1] (degrees)	Azimuth [2] (degrees)	Vehicle Weight (pounds)
1. Liftoff	0.00	28.65	-80.64	1,340	0	0.00		1,296.454
2. End Vertical Rise, Begin Zero-lift Flight	10.00	28.65	-80.64	1,345	510	4.53	44.0	1,232,181
3. Shutdown of S-IB Inboard Engines	134.72	28.98	-80.27	6,980	177,327	28.48	44.0	431,980
4. Shutdown of S-IB Outboard Engines (Burnout) Begin Coast	141.64	29.06	-80.19	7,357	200,629	27.36	44.0	410,107
5. Jettison of S-IB; S-IVB Ignition; Mixture Ratio = 5.0; Constant Attitude Rate	145.94	29.10	-80.14	7,295	214,935	26.49	44.1	304,281 [3]
6. Jettison Ullage Cases and Thermo Lag, Change Mixture Ratio to 5.5	155.94	29.21	-80.02	7,375	246,662	24.59	44.1	299,234 [4]
7. Jettison Launch Escape System	165.94	29.33	-79.89	7,495	276,624	22.76	44.2	285,598 [5]
8. Change Mixture Ratio to 4.7	440.00	34.32	-73.75	16,126	556,384	-1.37	48.5	136,617
9. Burnout S-IVB (80 N. Mi. Parking Orbit) End Integrated Trajectory	595.54	39.37	-65.93	25,665	486,057	0.00	53.4	67,391
10. Service Module Ignition for Perigee Maneuver	15:19:55 [10]	45.2	-31.8	25,665	80 N. Mi.	0.00		32,406 [6]
11. Service Module Burnout, Coast to Apogee [7]	15:20:07	45.2	-31.8	25,879	80 N. Mi.	0.00		31,659
12. Service Module Ignition for Apogee Maneuver	16:04:45	-45.2	136.6	25,016	200 N Mi.	0.00		30,659

Table 1-8

Saturn IB Reference Trajectories; 200-N. Mi. Circular Orbit by Hohmann Transfer
(Perigee = 80 N. Mi., i = 50.3) (Cont.)

Trajectory Event	Time (seconds)	Geodetic Latitude (degrees)	Longitude (degrees)	Velocity[1] (feet per second)	Altitude (feet)	Flight Path Angle[1] (degrees)	Azimuth[2] (degrees)	Vehicle Weight (pounds)
13. Service Module Burnout	16:04:56	-45.2	136.6	25,229	200 N. Mi.	0.00		30,929
14. Service Module Ignition for Deboost Maneuver	00:00:00	0.0	31.0	25,229	200 N. Mi.	0.00	140.3	20,664[8]
15. Service Module Burnout	00:00:11	0.0	31.0	24,870	200 N. Mi.	0.00	140.3	19,870
16. Command Module Reentry	00:27:39	-46.5	143.6	25,831	66 N. Mi.	-1.52	67.6	10,215

(1) Inertial Quantities.
(2) Relative Quantities.
(3) After Jettison Event of 105,826 pounds.
(4) After Jettison Event of 235 pounds.
(5) After Jettison Event of 8,220 pounds.
(6) After Jettison Event of 34,985 pounds.
(7) Each S/M Operation Provides an Addition 10 Percent Velocity Contingency over the Velocity Increment Required.
(8) The Difference between the Weights of Events 13 and 14 Represents Payload Left on Orbit or Expended During Stay.
(9) After Jettison Event of 10,410 pounds Which Represents the Spent S/M Plus 210 pounds (100 feet per second) for Midcourse Corrections.
(10) Allows 3 Hours 10 Minutes Parking Orbit Time.

CONFIDENTIAL

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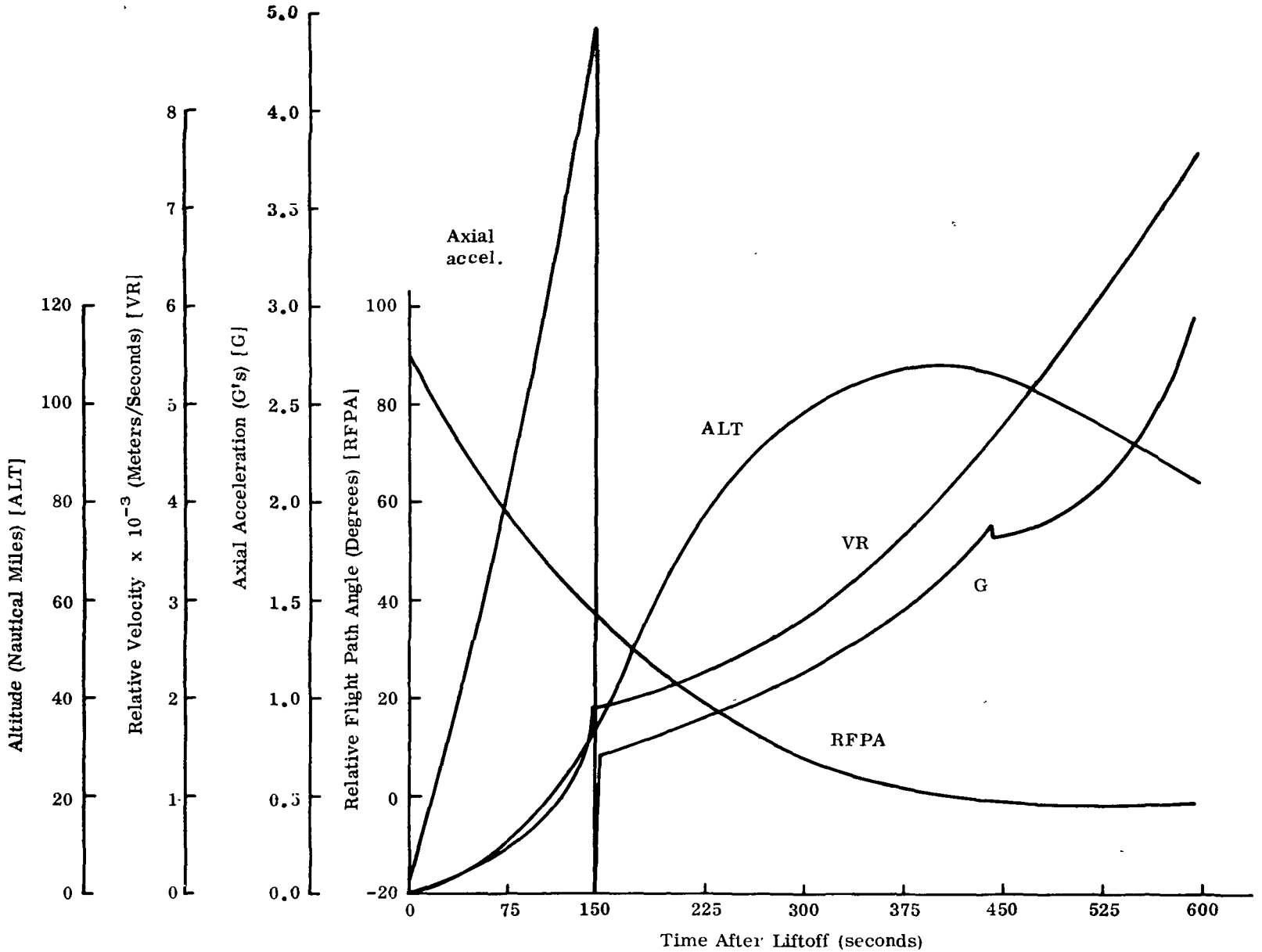


Figure 1-10. Axial Acceleration, Relative Flight Path Angle, Relative Velocity and Altitude as a Function of Time During Launch

1.2.6.2.2 Ground Rules

For all LES aborts, except those initiated during the first 40 seconds of flight, launch vehicle engine shutdown will precede LES ignition. During the first 40 seconds of flight, engine shutdown is inhibited by an AMR range safety constraint.

An automatic sequencing of spacecraft events is provided for aborts initiated below an altitude of 120,000 feet. Above 120,000 feet, the astronauts are required to jettison the LES manually, after escape motor thrust tail-off, and to orient the spacecraft to a heat-shield-forward attitude.

1.2.6.2.3 Mode Description

The operational events that follow the initiation of an LES abort, controlled by sequential timers and barometric switches, are shown in Figure 1-14. At abort initiation, launch vehicle engine shutdown is commanded (except during the first 40 seconds of flight). The launch escape and pitch control motors are then fired simultaneously. The pitch motor burns for approximately 0.5 second to provide lateral separation from the booster, while the escape motor thrusts for about 3.5 seconds to provide longitudinal separation.

At 11 seconds after abort, the canards are deployed. At 14 seconds after abort, if a 25,000-foot barometric switch (closed during prelaunch) is still closed, the tower jettison motor separates the LES, including the CM apex cover, from the CM. If the switch is open, LES jettison is delayed until reaching 25,000 feet, during descent, when the barometric switch is closed again. Drogue parachute deployment occurs two seconds after LES jettison. Main parachute deployment occurs two seconds after LES jettison. Main parachute deployment is controlled by a timer and barometric switch similar to LES jettison. At 12 seconds after drogue deployment, if a 12,000-foot barometric switch is still closed, the drogue chutes are disconnected and the main chutes are deployed. If the switch is open, the drogue chutes are retained until reaching 12,000 feet, on descent, when the switch is closed again.

If the abort occurs above an altitude of approximately 120,000 feet, the canard subsystem is ineffective and the pilot must manually orient the CM in a heat-shield-forward attitude. To do so, the LES must be manually jettisoned by the pilot after launch escape motor thrust tail-off (about 8 seconds). The CM apex cover is left on. Automatic RCS rate damping begins at tower jettison. After descending to 25,000 feet, the apex cover is jettisoned upon closing of the barometric switch, previously

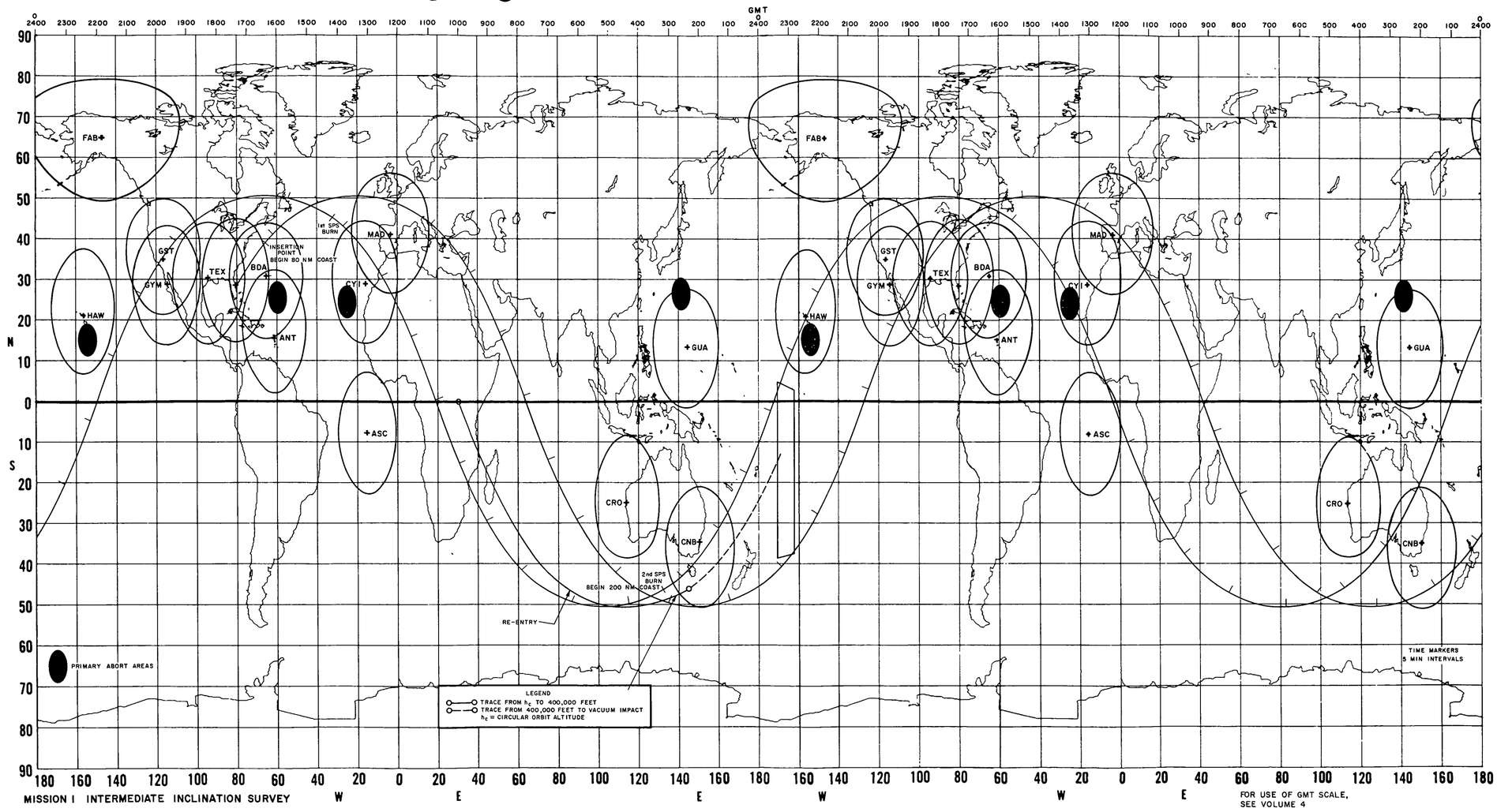


Figure 1-11. Mission 1 - Launch, Mission Orbit Insertion and Re-Entry Ground Trace

MISSION I - INTERMEDIATE INCLINATION SURVEY

ILLUMINATION AND RADAR COVERAGE CHART

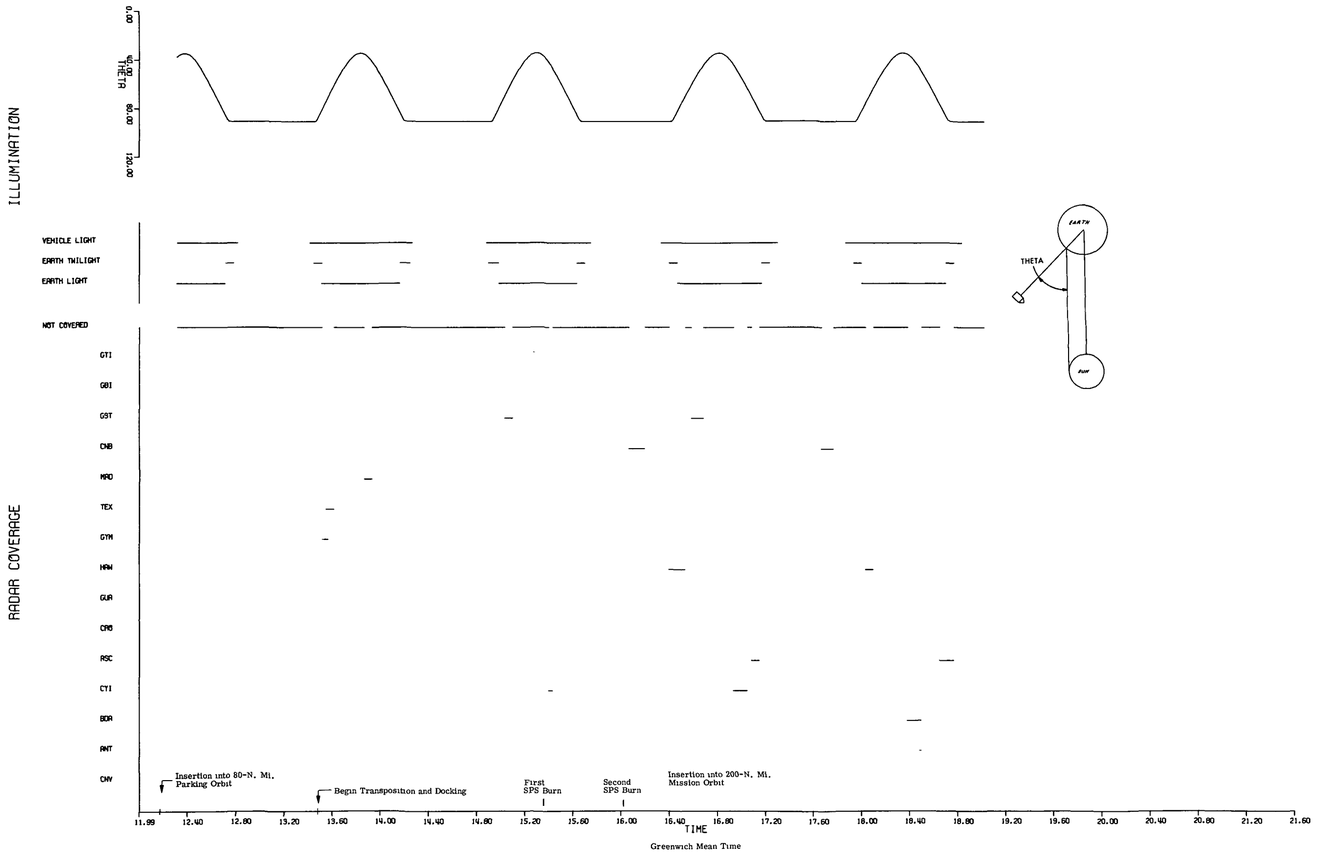
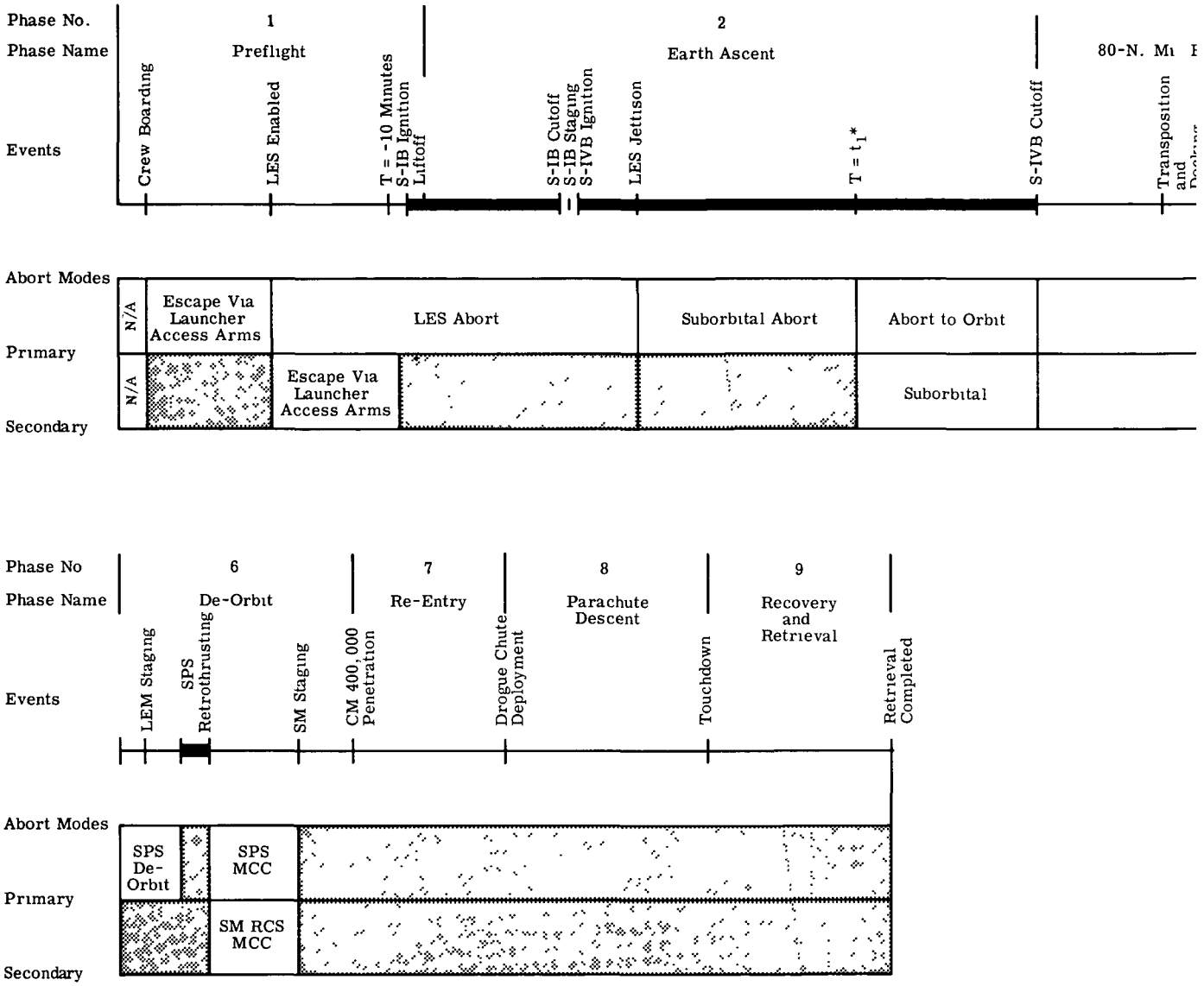


Figure 1-12. Illumination and Radar Coverage Chart - Mission 1



* t_1 is defined as that time at which the SPS is able to insert the CSM into the 80-n mi earth parking orbit

** t_2 is defined as that time at which the SM RCS is no longer capable of providing sufficient thrust to perform the de-orbit maneuver

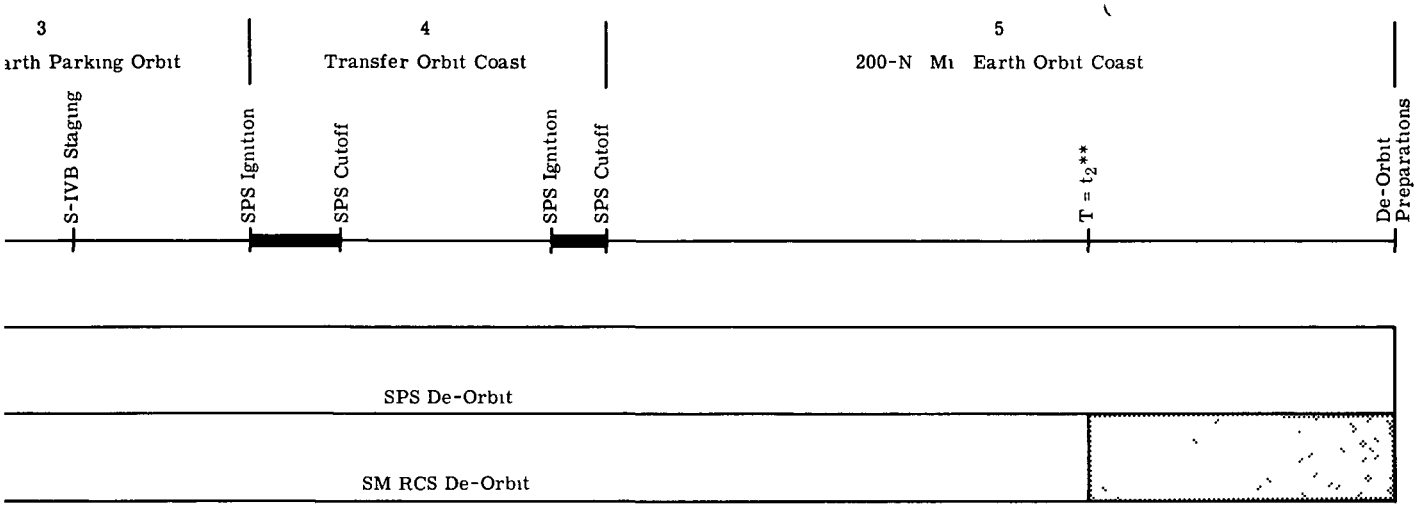


Figure 1-13. Mission Phase Versus Abort Mode - Mission 1

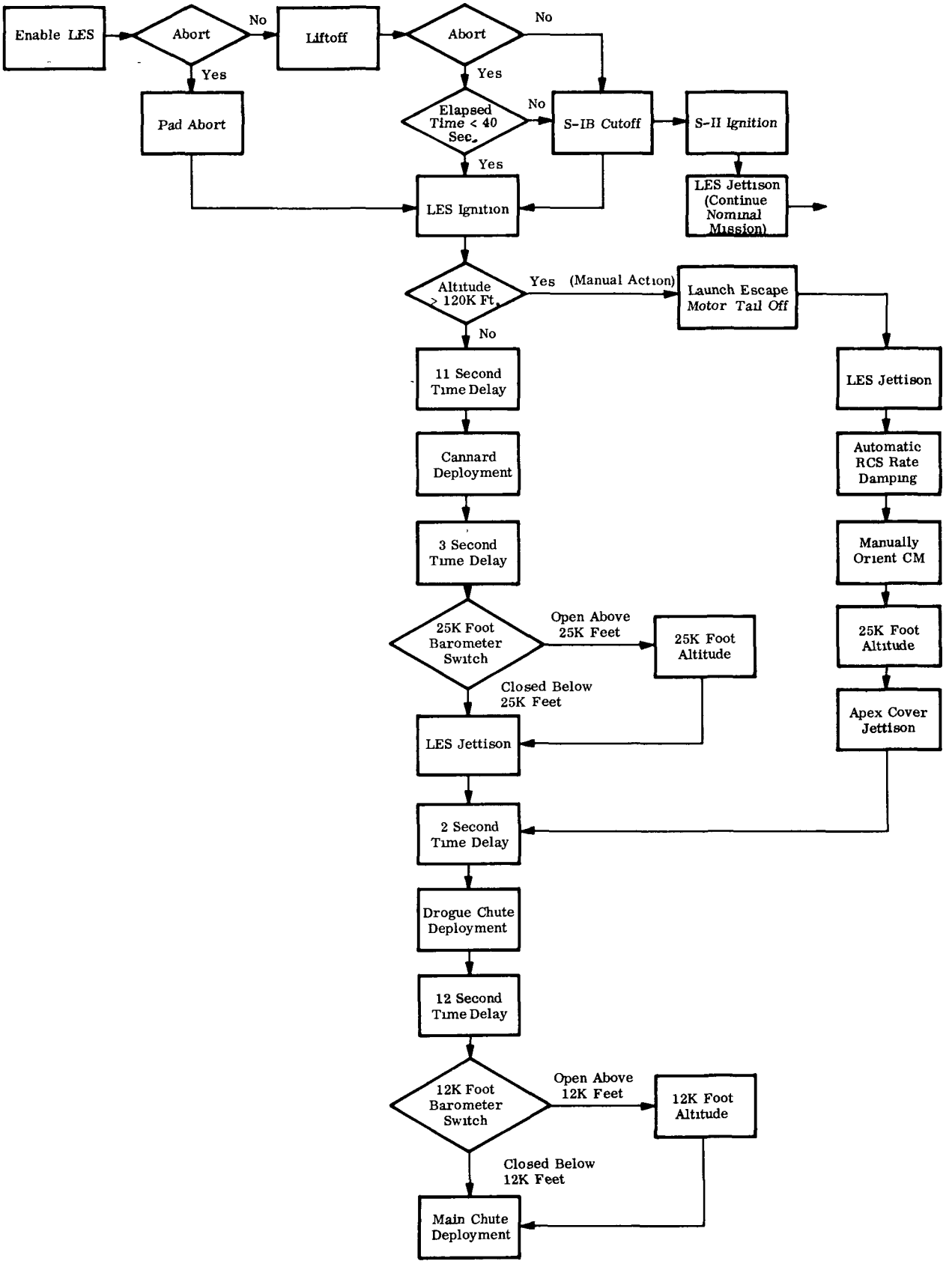


Figure 1-14. LES Abort Mission 1 Intermediate Inclination Survey

discussed. Two seconds later the drogue parachutes are deployed. At 12,000 feet another barometric switch closes, which disconnects the drogue chutes and deploys the main chutes.

1.2.6.3 Suborbital Aborts

1.2.6.3.1 Ground Rules

A suborbital abort may be performed any time between LES jettison and earth parking orbit insertion.

From LES jettison to t_1 second after liftoff, the suborbital abort is the primary abort mode. In this abort mode, the CM lands in the AMR. A suborbital abort is the secondary abort mode from t_1 seconds after liftoff to S-IVB cutoff at the completion of the earth parking orbit insertion maneuver. In this abort mode, the CM lands in the Indian Ocean recovery area.

Launch vehicle shutdown must precede initiation of suborbital abort action. The CSM abort separation procedure will utilize both an SM-RCS and an SPS burn of a fixed initiation time, ΔV, and attitude. The SM is separated from the CM at a fixed time after SM engine shutdown, following completion of the suborbital abort maneuvers.

The primary abort mode requires that the CM fly a maximum lift trajectory during entry to avoid excessive decelerations. The secondary abort mode requires that the CM fly a maximum lift trajectory during entry to avoid excessive decelerations. The secondary abort mode requires SPS posigrade thrust to attain the range of the landing site. A constant bank angle of approximately 60 degrees will be flown by the CM during entry.

1.2.6.3.2 Mode Description

The sequence of events that follow suborbital abort action is shown in Figure 1-15. In a suborbital abort, following booster shutdown, the SM-RCS is used to provide approximately 45 inches of separation between the CSM and launch vehicle. The SPS engine nozzle must be clear of the LEM SLA before the SPS can be ignited. It may be necessary to fire the SPS as soon as possible to damp out any existing pitch and yaw rates, which could exist at abort, because of a launch vehicle control system failure. After the SM RCS separation and SPS damping maneuvers are completed, a turn-around to the required thrust attitude is achieved by pulsing the SM-RCS.

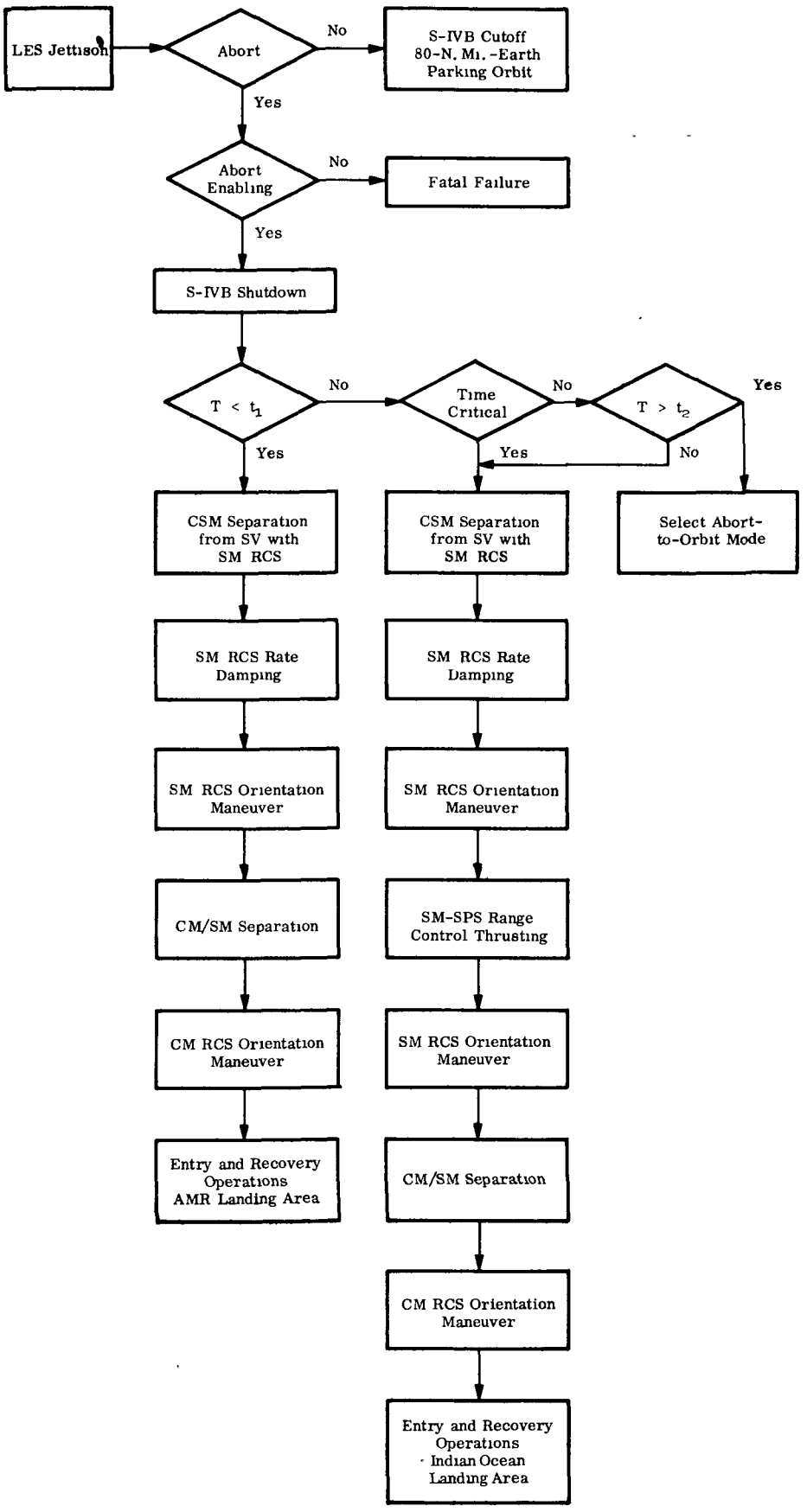


Figure 1-15. Suborbital Abort Mission 1 Intermediate Inclination Survey

Depending on the range control required, the SPS can be burned for variable lengths of time in either a posigrade or retrograde attitude. The time delay to thrust initiation also can be varied. After the CM is separated from the SM, the CM is capable of re-entering ballistically or with variable lift.

SPS range control thrusting is not used for aborts initiated during the suborbital primary mode (between LES jettison and t_1 second after liftoff). The CM, flying a maximum lift trajectory, will land in the AMR. Therefore, in the primary abort mode the SM-RCS establishes the correct CM/SM separation attitude after SM-RCS launch vehicle separation velocity. The CM has no translatory thrust capability. Following separation, the CM-RCS is used to orient the CM for a maximum lift entry.

From t_1 seconds after liftoff to S-IVB cutoff (the suborbital secondary mode), SPS posigrade range control thrusting is used to avoid a possible landing on the African continent, and ensure a CM landing in the Indian Ocean recovery area. Following SPS thrusting, the SM-RCS establishes the correct CM-SM separation, the CM-RCS orients the CM for a constant bank angle utilizing approximately half-lift capability.

1.2.6.4 Abort to Orbit

1.2.6.4.1 Ground Rules

An abort to orbit can be performed anytime from the point of orbiting capability of the CSM ($T = t_1$) to nominal S-IVB cutoff completing the parking orbit insertion maneuver.

An abort-to-orbit is the primary abort mode from t_1 seconds after liftoff (orbital capability) to S-IVB cutoff (at parking orbit insertion). The secondary abort mode for this period and all time-critical aborts are suborbital. S-IVB shutdown precedes the initiation of spacecraft abort action. The CSM separation procedure for an abort-to-abort will utilize both an SM-RCS and an SPS burn of a fixed initiation time ΔV and attitude.

1.2.6.4.2 Mode Description

The advisability of aborting to orbit once the capability exists, depends on the malfunction. If crew safety is not diminished, one advantage of aborting to orbit is that more flexibility in landing site selection is available. The sequence of events in an abort to orbit is shown in Figure 1-16.

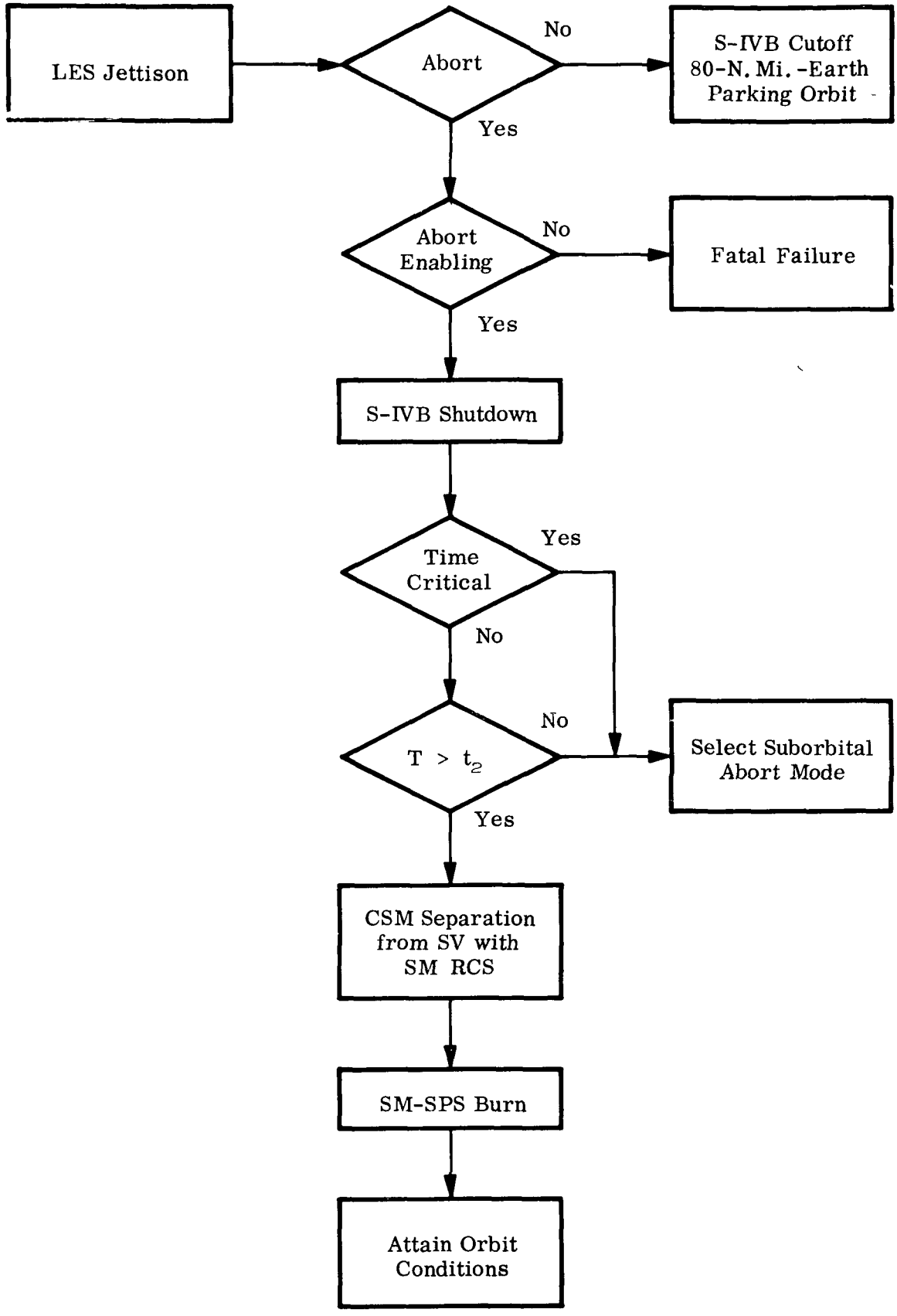


Figure 1-16. Abort-to-Orbit Mission 1 Intermediate Inclination Survey

Following S-IVB shutdown, the SM-RCS provides approximately 45 inches of separation between the CSM and the launch vehicle to clear the SPS engine nozzle of the SLA, a requirement which must be met before the SPS can be ignited. Following CSM separation, the SPS is ignited and thrusted until orbit insertion is achieved.

Sufficient SPS propellant must be reserved for the de-orbit maneuver. An SPS abort-to-orbit from an abort occuring at T = t_1 requires approximately ( ) seconds of SPS thrusting.

1.2.6.5 De-orbit Abort

1.2.6.5.1 Ground Rules

From S-IVB cutoff at earth orbit insertion and SPS ignition for the nominally scheduled de-orbit maneuver, the only abort possible is through an early de-orbit maneuver.

The primary abort mode utilizes the SPS to accomplish de-orbiting; the secondary abort mode utilizes the SM-RCS. The de-orbit maneuver is executed using a single propulsion burn at a fixed retrograde attitude and for a specified ΔV.

The spacecraft will consist of only the CSM at the time de-orbit thrusting is initiated. The CM/SM separation procedure will use a service module RCS translation and roll maneuver.

If time permits, the deorbit maneuver will be initiated at a time that permits entry to one of the planned recovery areas, preferably the primary recovery area.

1.2.6.5.2 Mode Descriptions

If a de-orbit abort is required before the completion of transposition and docking, the S-IVB-IU-SLA-LEM configuration is staged, whereas if transposition and docking is completed, only the LEM is staged. Following staging operations, the CSM is oriented for the de-orbit maneuver using the SM-RCS.

The SPS is the primary propulsive mode for the de-orbit abort. The SM-RCS serves as a secondary propulsive mode until a mission time to t_2, the time at which the SM-RCS propellant usage has depleted the ΔV capability to the point where it is no longer sufficient for the required de-orbit thrusting. If the de-orbit abort is required during any SPS thrusting period, the SPS must be shut down.

Time-critical aborts require immediate initiation of the de-orbit maneuver with minimum consideration given to the landing area. When the situation is not time-critical, the de-orbit maneuver can be delayed as necessary to land in a designated recovery area.

The sequence of events following the de-orbit abort thrusting maneuver is the same as events that follow the nominal de-orbit maneuver (Figure 1-17).

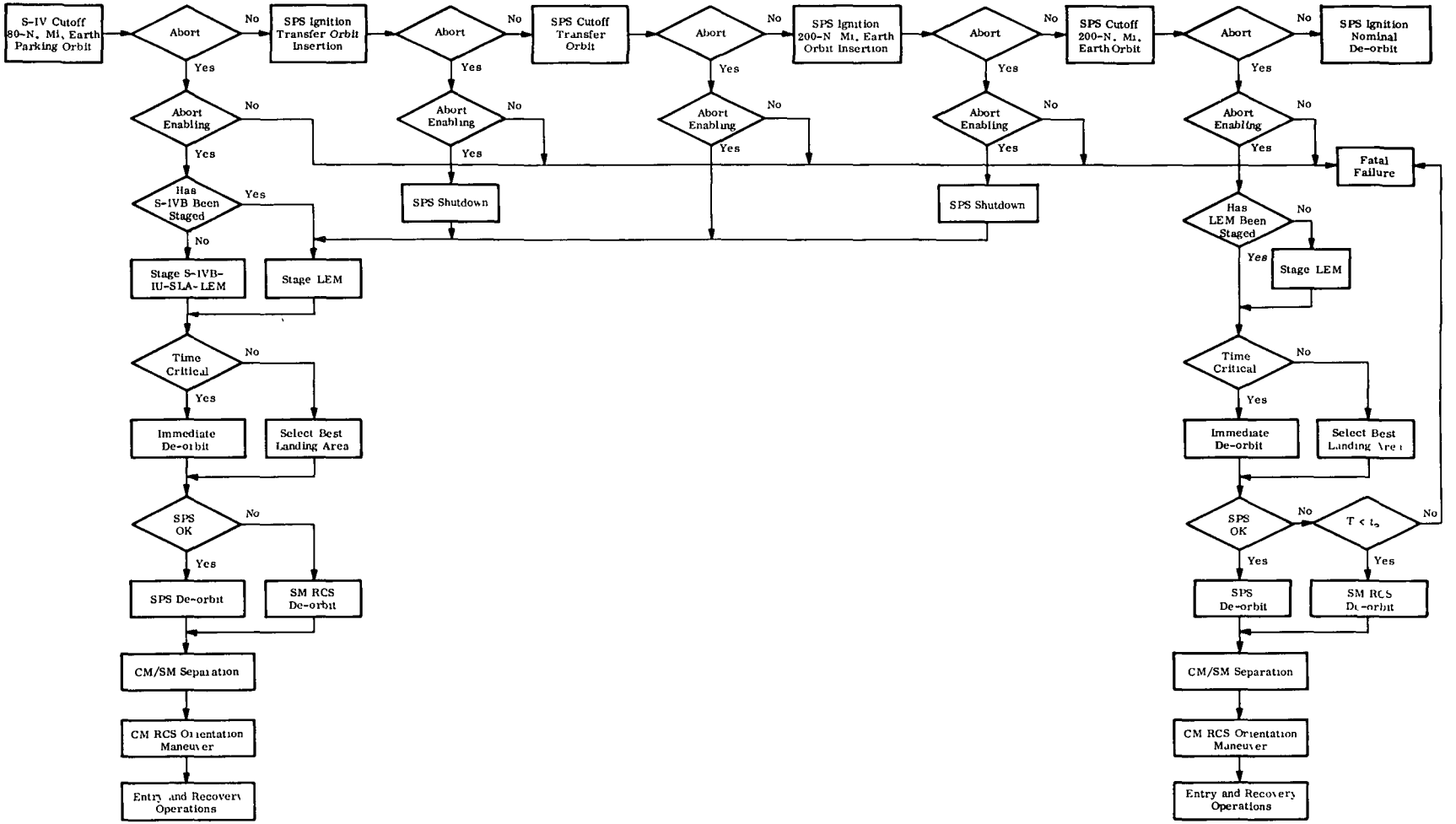


Figure 1-17. De-Orbit Abort Mission 1 Intermediate Inclination Survey

SECTION 2

MISSION 2

SATURN IB TO SATURN IB RENDEZVOUS

2.1 MISSION 2 DESCRIPTION

2.1.1 GENERAL

The Saturn IB to Saturn IB Rendezvous mission will obtain data in the bioscience and physical science fields, and continue experiments related to the development of space operations (see 1). The duration of Mission 2 is nominally 90 days (see 19).

2.1.2 MISSION SUMMARY

2.1.2.1 Introduction

The first spacecraft, "A," consisting of a basic Apollo Block II CSM and a modified LEM, will be placed in a nominal 200-n. mi. circular earth orbit at an inclination of 29.5 degrees (see 50). The spacecraft will carry a crew of three, and mission expendables and subsystems that will permit operations for not less than 45 days in earth orbit (see 1).

A second spacecraft, "B," of the same configuration and crew size as Spacecraft A, will be placed in a similar orbit 35 to 45 days after the launch of Spacecraft A (see 19). The orbit of Spacecraft B will permit rendezvous and docking with Spacecraft A. After docking, and transfer of scheduled personnel, equipment, and other supplies, the CSM of Spacecraft A will separate from the LEM A and Spacecraft B, de-orbit, and return to earth for recovery. Spacecraft B will remain in orbit for the remaining 44 days of the mission; then it will de-orbit and return to earth for recovery (see 17).

2.1.2.2 Configuration

The launch vehicles for Mission 2 will be the Saturn IB launch vehicle. The Saturn IB is a multistage launch vehicle, consisting of two propulsive stages (S-IB and S-IVB) and a nonpropulsive module, the Instrument Unit (IU) (see 101-6).

The spacecraft for each flight of Mission 2 is a modified Block II Command and Service Module (CSM), a modified Lunar Excursion Module (LEM/LAB), a Launch Escape System (LES), and a Spacecraft LEM Adapter (SLA), with added expendables and subsystems to permit operations of up to 45 days in earth orbit (see 101-2).

2.1.2.3 Experiments

2.1.2.3.1 First Flight

Sixteen types of experiments in the bioscience-physical science fields are planned for this flight (see 35-31). The major objective is to obtain the initial data necessary to establish a space laboratory. A substantial portion of the total experiment weight (approximately 50 percent) is in evaluation experiments of spacesuit assemblies (see 31.1.7).

The total weight of the equipment required for experiments is approximately 2600 pounds. It will occupy 156 cubic feet and require an average power of 245 watts, with peaks to 1350 watts (see 35.3.1).

2.1.2.3.2 Second Flight

There are seven types of experiments in the bioscience and space operations fields planned for the second flight. A substantial portion of the total experiment weight (approximately 43 percent) is required for the onboard centrifuge. These experiments are performed while hard docked to LEM A, which carries the experiment equipment of the first flight.

The total weight of experiment equipment is approximately 2200 pounds. It will occupy 486 cubic feet and require an average power of 352 watts, with peaks to 1500 watts.

2.1.2.4 Operations

Spacecraft A, utilizing the Saturn IB launch vehicle, will be launched from Cape Kennedy Launch Complex 34 on a launch azimuth of 83 degrees into a nominal 80-n. mi. circular earth parking orbit. The inclination of the parking orbit will be 29.5 degrees (see 50).

Transposition and docking is accomplished during the parking orbit. The S-IVB, including the IU and SLA, is staged soon after a hard dock is attained between the CSM

and LEM. The final portion of the parking orbit phase is used for preparations for SPS transfer orbit insertion.

Upon completion of transfer orbit insertion preparations, the SPS inserts Spacecraft A into a co-planar Hohmann transfer orbit with an apogee of 200 n. mi. Spacecraft A coasts in the transfer orbit until apogee, at which time the SPS is fired a second time in order to place the spacecraft in a circular orbit at the nominal 200-n. mi. mission altitude. At SPS cutoff the spacecraft is in the 200-n. mi. circular earth orbit; the orbit inclination is equal to that of the mission orbit. After the post-insertion tasks are completed, the interlock and LEM are pressurized and manned, and upon completion of the required LEM subsystem checks, the mission's prime experiments are initiated.

The astronauts begin a work/rest cycle. They perform, adjust, and calibrate any experiments or associated apparatus, perform station-keeping duties, and accomplish their normal spacecraft functions until several orbits prior to the launch of Spacecraft B on or before the forty-fourth mission day. The launch of Vehicle B may be scheduled as early as the thirty-fifth day to allow for any launch-date slippage. Mission times would advance accordingly.

On or before the forty-fourth mission day, Spacecraft B will be launched from Cape Kennedy Launch Complex 37B on a launch azimuth based on mission requirements, into a nominal 80-n. mi. circular earth-parking orbit by the Saturn IB launch vehicle. The timing of the Saturn IB launch will permit Spacecraft B to transfer from the parking orbit during the third orbit, and rendezvous with Spacecraft A at the 200-n. mi. mission altitude. The inclination of the parking orbit will be based on mission requirements, as is the launch azimuth, but will be between 28.5 and 31.5 degrees, based on a plane change capability of one degree (see 50). This one-degree plane change capability is critically dependent on orbital rendezvous fuel requirements (Part II, Section 5).

Vehicle B accomplishes transposition and docking during the parking orbit. Shortly after a hard dock is made between CSM B and LEM B, the B vehicle's S-IVB stage, including the IU and SLA, is staged. At the correct phasing point, the SPS inserts Spacecraft B into a co-planar Hohmann transfer orbit having an apogee of 200 n. mi. Spacecraft A and Spacecraft B track each other during Spacecraft B's ascent coast. Spacecraft B performs midcourse corrections as required during the transfer orbit ascent, then begins preparations for terminal rendezvous maneuvers, which occur

during the later part of the ascent coast. Rendezvous maneuvers are accomplished by Spacecraft B according to a prescheduled plan, based on the line-of-sight range and the range rate between the two vehicles.

Upon completion of the terminal rendezvous maneuvers, the two spacecraft are within a specified distance of one another. At this time, LEM A separates from CSM A and accomplishes a hard dock with LEM B, which remains docked to CSM B. Following completion of LEM A docking to Spacecraft B, CSM A accomplishes a hard dock to LEM A. At this point, the two spacecraft are in a LEM-to-LEM docked configuration. This configuration is maintained until the planned crew, supply, and material transfers are complete. After preparations are completed, nominally a period of one day, the CSM A crew effects a separation from the CSM B, LEM A and B. Following separation, CSM A de-orbits, re-enters, and is recovered. CSM B and LEM A and B continue in the mission orbit for the following 44 days.

2.1.3 MISSION OBJECTIVES

2.1.3.1 Experiment Objectives

2.1.3.1.1 Spacecraft A

The experiment objectives for the first Spacecraft A flight include the study of living organisms in a zero-g environment, and a rendezvous of this flight with a second spacecraft for extended study of this and other areas, such as the behavior of liquid gas and solids, and space environment studies. Experiments (see 35-5) include the following:

a. 0100 Biomedical (0101 and 0102 performed during rendezvous flight).
b. 0200 Behavioral.
c. 0500 Radiation Environment Monitoring.
d. 0601 Capillarity Studies.
e. 0602A Kinetics and Dynamics of Vapor/Gas Bubbles.
f. 0602B Liquid Drop Dynamic Studies.
g. 0603A Pool Boiling in Long-Term Zero-g.
h. 0603B Nucleate Condensation of Fluids in Zero-g.
i. 0604 Density Gradient of a Fluid.
j. 0605 Crystallization Studies.
k. 0606 Cosmic Ray Emission (performed during rendezvous flight).
l. 1501 Evaluation of Advanced Spacesuit Assemblies.

m. 1502 Development of Manned Locomotion and Maneuvering Capability.
n. 1503 Emergency Techniques for Rescue.
o. 1504 Development of Personnel and Cargo Transfer Operations.
p. 1505 Maintenance and Repair.

2.1.3.1.2 Spacecraft B

The experiment objectives for this flight include a rendezvous with Spacecraft A to provide continuation of living organisms studies; physical science experiments and space environment studies; further investigation of extravehicular operations, including cargo and personnel transfer docking, rescue operations, and spacesuit evaluations; and the effects of space flight on biological systems, microbes, and mammals.

Experiments (see 35-9) include the following:

a. 0302 Onboard Centrifuge.
b. 0401 Microbial Genetics and Growth.
c. 0402 Development Biology.
d. 0403 Mammalian Physiology.
e. 0404 Limb Regeneration and Wound Healing.
f. 0405 The Effects of Drugs on Mammalian Behavior.
g. 0406 Soft Capture Enumeration, and Identification of Spaceborne Microorganisms.

2.1.3.2 Flight Systems Objectives

The objectives of the two Saturn IB launch vehicles are to place the two spacecraft payloads of this mission into an 80-n. mi. parking orbit.

The spacecraft objectives are to carry the mission experiment equipment and a crew of three to the specified mission orbit, effect rendezvous of the spacecraft payloads to yield the second flight spacecraft configuration of a CSM and two LEM's, effect transfer of crew personnel and material, return the crews and data samples from each flight, and provide operational support for the crew and experiments during each of the two 45-day flights.

The crew, for each of the two flights that comprise this mission type, will consist of three astronauts (see 9-4), who are expected to participate in the performance of experiments beyond merely checking out and operating equipment. They will help to insure the scientific and technical validity of the experiments and provide partial

evaluations of results (see 13). One crew member will remain aloft 90 days (see 48). The technical specialty of each crew member, and the determination of which crew member will be exchanged after the rendezvous operations, has not been defined.

2.1.3.3 Ground Support Objectives

2.1.3.3.1 Mission Command Control

The mission command responsibility will remain the same as for the basic Apollo mission, with provisions for handling two vehicles in flight simultaneously, and the ability to pass the mission command to the onboard crew commander for all but the critical mission phases (see 15).

2.1.3.3.2 Launch Support and Control

The normal complement of equipment at Pads 34 and 37B (see 7) will be utilized with the Launch Control Center for each of the two flights comprising this mission. Checkout operations and modifications to launch facilities will be minimized (see 9-13).

Assurance that the second Saturn IB launch will occur 35 to 45 days following the first Saturn IB launch is a constraint necessary to achieve all the experiment objectives.

2.1.3.3.3 Tracking and Communications

All MSFN remote sites supporting this mission will communicate directly with MCC Houston, with sufficient data routed in real-time to preclude the necessity for remote site controllers (see 15-11). Tracking and communications are required continuously throughout Saturn IB powered flights (see 15-66). Provisions to communicate and track both spacecraft simultaneously, when in near proximity of each other, from a single MSFN site, is an objective that requires consideration of remote-site frequency assignments and system coverage.

2.1.3.3.4 Recovery

Mission recovery operations will be similar to those of the basic Apollo system, with the Pacific Recovery Area remaining the primary recovery area (see 24). The potential use of recovery rescue aircraft will be considered in lieu of a continuous assignment of ships for worldwide coverage of abort conditions (see 9-70). When both vehicles are in flight, the ability to effect recovery of both vehicles during a 1-to-3 day period must be considered.

2.1.3.3.5 Post-Flight

The objectives for post-flight operations, involving such activities as data reduction, equipment performance review, and ground support performance reviews have not been considered for AAM program documentation.

2.1.4 MISSION GROUND RULES

2.1.4.1 General

The following set of ground rules is intended to supply guidance for the profiling of Mission 2, Flights 218 and 219.

2.1.4.2 Earth Launch Phase

The Merritt Island Launch Area (MILA) launch pad assignment for the launch of Flight 218 is Pad 34. The MILA launch pad assignment for the launch of Flight 219 is Pad 37B (see 5). The launch windows include the following:

a. Launch azimuth for the first flight of this mission will be 83 degrees (see 2.2.5).
b. Range safety launch azimuth limits will be 72 to 108 degrees, based on the earth-fixed heading of the launch vehicle at the beginning of the pitch program (see 29-1).
c. Operational launch azimuth limits will be limited to a range of 26 degrees (see 29-1).

Continuous tracking is required from earth launch to three minutes after insertion into the initial orbit (see 91).

2.1.4.3 Earth Parking Orbit Phase

The nominal earth parking orbit will be a near-circular orbit of 80-n. mi. altitude (see 19).

Transfer orbit injection will occur at the earliest opportunity after spacecraft systems have been checked and the orbit ephemeris determined (see 29-2), but injection will occur not later than three orbits after earth parking orbit insertion (see 47-2). The third orbit is considered the nominal injection orbit (see 2.2.5).

MSFN will be the primary source for trajectory determinations (see 26-5). A minimum of two station contacts is required during the first parking orbit, and one station contact during all subsequent parking orbits (see 14-33). For station contact consideration, a contact time of at least four minutes at an elevation angle greater than 5 degrees is required (see 91).

The astronauts will have control of the S-IVB reaction jets for attitude control from earth orbit insertion to separation of the CSM from the S-IVB/LEM (see 24-2). There will be no natural lighting constraints on transposition and docking (see 29-3) other than the requirements for sun illumination of the LEM docking tunnel and the sun vector angle of less than 90 degrees with the CSM X-body axis. No ground rules on communications requirements during transposition and docking can be written at this time.

2.1.4.4 Transfer Trajectory Phase

The transfer trajectory from the 80-n. mi. earth parking orbit to the 200-n. mi. mission orbit will be a Hohmann ellipse (see 19).

One 4-minute preburn contact with the MSFN within 90 minutes before burn is required (see 91). One 4-minute post-burn contact is required not later than 7 minutes after burn (see 19).

2.1.4.5 Mission Orbit Phase

The nominal mission orbit will be a near-circular orbit of 200 n. mi. altitude (see 19). Orbit inclination will be 29.5 degrees (see 50-7).

The flight mission is under control of the flight director, who will, under normal circumstances, delegate control of the scientific program when the spacecraft has achieved its design and flight path (see 14-55). One 4-minute contact per orbit with MSFN is required (see 91). The elapsed time between two successive contacts should not exceed two hours (see 91).

The design of the spacecraft will permit extravehicular transfer of the crew between CM and LEM in either direction, in both the docked and undocked conditions (see 24-3). Docking maneuvers will be manually controlled (24-2). In all earth orbital missions, the SPS will be used for de-orbiting (see 14-63).

No more than two astronauts will be off duty at any one time during the mission (see 76-5). At least one astronaut will be wearing a spacesuit at all times during flight.

The faceplate may be open or closed (see 24-2). One crew member will be exchanged from Spacecraft A to Spacecraft B (see 47).

Experiment scheduling details are provided in paragraph 2.2.3 of this report.

The following rendezvous ground rules are proposed based on results of this mission analysis:

a. The primary command, control, and computation of course rendezvous maneuvers will be the responsibility of MCC.
b. The altitude of Vehicle A will be maintained at 200 n. mi. The timing of any altitude restoration maneuvers will be used to adjust Vehicle A orbit phasing.
c. The primary responsibility for terminal rendezvous maneuvers will rest with the commander of Vehicle B.
d. Vehicle B liftoff must occur within a launch window of 24 minutes.

The LEM/LAB and CSM will dock on the supply spacecraft separately, and in the order, LEM/LAB CSM (see 38-8). Coast duration in the docked configuration will be a maximum of 24 hours (see 48). Data is not available on equipment and data transfer requirements at this time.

2.1.4.6 Re-Entry Phase

Trajectory characteristics of the re-entry phase are as follows:

a. Vacuum perigee equals 0 n. mi. (see 49-2).
b. Re-entry corridor - not defined at this time (see 29-7).
c. Re-entry range - not defined at this time (see 29-7).
d. Equatorial inclination of the inertial velocity vector during re-entry will not be greater than ( ) degrees. The re-entry trajectory will be posigrade.
e. Cross-range maneuvering is not required for the nominal entry trajectory (see 24-5).

One four-minute contact is required with MSFN within 90 minutes preceding retrofiring (see 91). One four-minute contact is required not later than seven minutes after retrofiring (see 91).

The primary recovery area is designated as the Pacific Recovery Area (see 29-9).

2.1.5 WEIGHTS SUMMARY

The spacecraft weights for Flight A (218) of Mission 2 are summarized in Table 2-1; spacecraft weights for Flight B (219) are summarized in Table 2-2. The tables are divided into three sections: Command Module, Service Module, and LEM/LAB. The major categories for each module are listed and referenced.

Each of the two command modules is a NAA Block II Mod D configuration (see 45). The service module for each flight of the two flights is a NAA Block II Mod D configuration (see 45), except for (1) RCS fuel loading, which is based on Grumman AES studies (see 32), and (2) the SPS retrofuel load, which is based on computations done by General Electric, assuming impulse burns and an additional 10 percent ΔV margin included in the retrofuel budget (see Part II, Section 6). The LEM/LAB weights are based on Grumman Aircraft and Engineering Corporation (GAEC) experiment assignments (see 32) as altered by Bellcomm (see 35) to meet margin requirements.

2.1.6 EXPERIMENT LOAD DESCRIPTIONS

2.1.6.1 Loads

The experiment loads for Mission 2 are shown in Tables 2-3 (Flight 218) and 2-4 (Flight 219).

The total weight of the experiment equipment for the first flight is approximately 2600 pounds, of which more than half is for the evaluation of spacesuit assemblies.

In the second Mission 2 flight (219), the total weight of the experiment equipment is approximately 2200 pounds, of which more than half is devoted to the onboard centrifuge equipment.

2.1.6.2 Support

The biomedical and behavioral experiments and radiation environment monitoring are planned for all earth orbital missions and have been discussed under Mission 1 (see 1.1.5).

For the experiments on the generic effects of microorganisms, identical control experiments will be conducted simultaneously in flight and in an earth laboratory. This support requirement applies to each of the experiments involving studies of living organisms, except the soft capture of spaceborne microorganisms.

Table 2-1

Mission Orbit Injection Weights Summary - Mission 2, Flight A (218)

Module		Weight (Pounds)	Reference No. and Page
1. Command Module			
Empty Weight		9,034	(45) No Page
Useful Weight		1,269	(45) No Page
	Total	[10,303]	
2. Service Module			
Empty Weight		8,921	(45) No Page
Reaction Control Propellant[1]			See Vol. 1
Experiments RCS Propellant			Part II
Excess RCS Propellant			
Cryogenic Expendables		653	(45) No Page
SPS Retrofuel[2]		810	Part II, Section 6
	Total	[10,384]	
3. LEM/LAB			
Structure		2,188	(32) 1.7.27
SCS		43	(32) 1.7.27
Crew Provisions		291	(32) 1.7.27
RCS		1,120	(32) 1.7.27
(Propellant)[3]		(620)	(32) 1.7.27
Controls and Displays		130	(32) 1.7.27
Environment Control		792	(32) 1.7.27
Instrumentation		301	(32) 1.7.27
EPS		4,333	(32) 1.7.27
(Reactant and O_2)		(2,230)	(32) 1.7.28
Communications		58	(32) 1.7.28
Experiments[4]		2,652	(35) para. 3.1
Spares		60	(32) 1.7.28
	Total	[11,968]	
4. Total Spacecraft Weight[5]		32,655	
Maximum Spacecraft Weight into Orbit[6]		34,685	(49) Page 10
Margin		2,030	

1. SM RCS Tank Capacity - 2,457 pounds (45)
2. This figure includes a 10 percent ΔV reserve.
3. The LEM/LAB RCS propellant load represents a deletion of 900 pounds from the GAEC total experiment RCS propellant requirements. This weight was reassigned to Flight 518.
4. The experiments package weight represents a reassignment of experiment series 1500 (weight 1,720) to Flight 218 from Flight 219 and reassignment of experiment 1407 (weight 369 pounds) from Flight 218 to Flight 518.
5. Summation of the bracketed numbers.
6. Maximum weight in orbit was increased by 1,500 pounds over the value cited in 49-10 (see 130).

Table 2-2

Mission Orbit Injection Weights Summary - Mission 2, Flight B (219)

Module		Weight (Pounds)	Reference No. and Page
1. Command Module			
Empty Weight		9,034	(45) No Page
Useful Weight		1,269	(45) No Page
	Total	[10,303]	
2. Service Module			
Empty Weight		8,921	(45) No Page
Reaction Control Propellant[1]			See Vol. 1
Experiments RCS Propellant			Part II
Excess RCS Propellant			
Cryogenic Expendables		653	(45) No Page
SPS Retrofuel[2]		824	Part II, Section 6
SPS Plane Change Fuel		455	2.2.5
	Total	[10,853]	
3. LEM/LAB			
Structure		2,134	(32) 1.8.29
Crew Provisions		350	(32) 1.8.29
RCS		720	(32) 1.8.29
(Propellant)[3]		(220)	(35) para. 4.1
Controls and Displays		387	(32) 1.8.29
ECS		979	(32) 1.8.29
Instrumentation		151	(32) 1.8.29
EPS		5,036	(32) 1.8.30
(Reactant and O_2)[4]		(2,638)	(35) para. 4.1
Communications		58	(32) 1.8.30
Experiments[5]		2,241	(35) para. 4.1
Spares		60	(32) 1.8.30
	Total	[12,116]	
4. Total Spacecraft Weight[6]		33,272	
Maximum Spacecraft Weight into Orbit[7]		34,685	(49) Page 10
Margin		1,413	

1. SM RCS tank capacity - 2,457 pounds.
2. This figure includes a 10 percent ΔV reserve.
3. The LEM/LAB RCS propellant loading represents a deletion of 1,610 pounds from the total GAEC experiment RCS requirements. The 1,610 pounds was reassigned to Flight 518.
4. The LEM/LAB EPS reactants and O_2 load represents a deletion of 412 pounds from the GAEC LEM/LAB EPS reactants and O_2 load. This weight was not reassigned to another flight.
5. The experiment package weight represents reassignment of Experiment Series 1500 (weight 1,720 pounds) from Flight 219 to Flight 218.
6. Summation of all bracketed numbers.
7. Maximum weight in orbit was increased by 1,500 pounds over the value cited in 49-10 (see 130).

Table 2-3

Experiment Load Summary, Flight A

(35-5) (35-5) (35-5) (35-7)

Experiment Number	Experiment Category/ Abbreviated Title	Total Weight (Lbs.)	Volume ($Ft.^3$)	Volume and Weight References	Power (Watts)	
					Average	Peak
0100	Biomedical (0101 and 0102 performed during rendezvous flight)	313	21.5	(32-1-7.5 & .6)	765.5	
0200	Behavioral	20	2.3	(32-1-7.7)	80	100
0501	Radiation Environment Monitoring	46.5	2.64	(32-1-7.7)	6.3	
0601	Capillarity Studies	113	3.63	(32-1-7.7)	430	
0602A	Kinetics and Dynamics of Vapor/Gas	41	0.336	(32-1-7.8)	100	
0602B	Liquid Drop Dynamic Studies	71	1.434	(32-1-7.8)	110	1350
0603A	Pool Boiling in Long-Term Zero-g	52.3	2.57	(32-1-7.8)	180	550
0603B	Nucleate Condensation of Fluids in Zero-g	21	1.27	(32-1-7.8)		550
0604	Density Gradient of a Fluid	42.1	2.01	(32-1-7.8)	200	
0605	Crystallization Studies	152.1	30.3	(32-1-7.8)	500	
0606	Cosmic Ray Emission (performed during rendezvous flight)	60	4.75	(32-1-7.8)	60	100
1501	Evaluation of Advanced Space-suit Assemblies	1155	37.6	(32-1-8.8 & .9)	5.0	
1502	Development of Manned Locomotion and Maneuvering Capability	364	18	(32-1-8.9)		
1503	Emergency Techniques for Rescue	68	15.1	(32-1-8.9)		
1504	Development of Personnel and Cargo Transfer Operations	100	8	(32-1-8.9)	250	
1505	Maintenance and Repair	34	5	(32-1-8.9)		
	Totals	2652	156.440		245	NA
	S/C Capability					

Table 2-4

Experiment Load Summary, Flight B

(35-9) (35-9) (35-9) (35-10)

Experiment Number	Experiment Category/ Abbreviated Title	Total Weight (Lbs.)	Volume ($Ft.^3$)	Volume and Weight References	Power (Watts)	
					Average	Peak
0302	Onboard Centrifuge	943	416.3	(32-1-8.9)	20	1500
0401	Microbial Genetics and Growth	60	7.7	(32-1-8.6)	25	85
0402	Development Biology	175	21.5	(32-1-8.6)	340	
0403	Mammalian Physiology	236	16.1	(32-1-8.7)	30	
0404	Limb Regeneration and Wound Healing	78	5.7	(32-1-8.7)	90	
0405	Effects of Drugs on Mammalian Behavior	708	17.6	(32-1-8.8)	56	
0406	Soft Capture, Enumeration, and Identification of Spaceborne Microorganisms	41	2	(32-1-8.8)	110	25
	Totals	2241	486.9		352	NA

S/C Capability

The five extravehicular activities (EVA) experiments will provide the first opportunity for personnel to participate in rendezvous and crew transfer. The transfer of personnel and supplies at the time of rendezvous, and subsequent extended tests with tether lines, mechanical systems and powered astronaut mobile unit operations, will require normal space flight ground support. For most EVA experiments, two crewmen are required, one outside the spacecraft and the second observing from the inside.

Spacesuit tests will involve manned operation of the suits in many work environments. Use of inflatable airlocks and evaluations of various failure detection devices will be part of the emergency rescue experiments. Maintenance and repair experiments will entail the application of various tools and repair techniques to tasks performed inside the airlock (with and without pressure suits) and outside the airlock. In the onboard centrifuge experiments, the objective is to determine the g-loading or exercise required to maintain physical fitness. One crewman is required in this experiment and will be subjected to periodic g-conditioning. Tolerances will be related to the results of earth laboratory experiments.

2.2 OPERATIONS

2.2.1 MISSION PHASE DESCRIPTIONS

2.2.1.1 Introduction

For the purpose of analysis and description, Mission 2 consists of 20 major phases. Table 2-5 shows each major phase, its initiating event, the time of the concluding event, and the time duration of each phase. The phase descriptions are detailed descriptions of operations and crew activities during each phase. As with Mission 1, there are many parameters that currently do not have values assigned. These parameters were included, based on comparable work in the AMPTF Design Reference Mission (see 24, 25, 26), in order to establish the basis for future development of mission phase descriptions.

The detailed phase descriptions* for each phase, including crew operations, are developed in the following paragraphs. The descriptions include crew operations. Appendix C identifies the crew operations adapted from AMPTF information.

*This description is based on References 24, 26, and 29 and the summary chart of the AAM Earth Orbit Missions, Table 1 of this report.

Table 2-5

Mission 2, Saturn IB to Saturn IB Rendezvous Mission Phase

Phase	Initiating Event	Event Time Hrs Min Sec	Concluding Event	Duration ΔT Hrs Min Sec
1.0 Preflight	T-450 hrs	T-450 hrs	Liftoff	450 hrs
2.0 Vehicle A Earth Ascent	Liftoff	0	Parking Orbit Insertion - S-IVB Cutoff	595.54 sec(1)
2.1 S-IB Thrusting	Liftoff	0	S-IB Staging	145.94 sec(1)
2.1.1 Coast		141.64		4.3 sec(1)
2.1.2 S-IB Staging		145.94 sec(1)		
2.2 S-IVB Thrusting	S-IB Staging	145.94 sec(1)	Parking Orbit Insertion - S-IVB Cutoff	449.60 sec(1)
2.2.1 LES Jettison		165.94 sec(1)		
3.0 Vehicle A 80-N. Mi. Earth Parking Orbit	Parking Orbit Insertion - S-IVB Cutoff	9 m, 55.54 s	TOI - SPS Ignition	3 h, 00 m (2)
3.1 Initial Coast	Parking Orbit Insertion - S-IVB Cutoff		CSM/LEM - S-IVB Separation	
3.2 Transposition and Docking	CSM/LEM - S-IVB Separation		Hard Dock	40 min
3.3 S-IVB/IU/SLA Staging	Hard Dock		TOI - SPS Ignition	

Table 2-5

Mission 2, Saturn IB to Saturn IB Rendezvous Mission Phase (Cont.)

Phase	Initiating Event	Event Time Hrs Min Sec	Concluding Event	Duration ΔT Hrs Min Sec
4.0 Vehicle A Transfer Orbit	TOI - SPS Ignition	3 h, 10 m	EOI - SPS Cutoff	45 min
4.1 TOI - SPS Thrusting	TOI - SPS Ignition	3 h, 10 m	TOI - SPS Cutoff	11.51 sec
4.2 Transfer Orbit Coast	TOI - SPS Cutoff		EOI - SPS Ignition	44 m, 37 sec
4.3 EOI - SPS Thrusting	EOI - SPS Ignition	3 h, 55 min	EOI - SPS Cutoff	11.24 sec
5.0 Vehicle A 200-N. Mi. Earth Orbit Coast	EOI - SPS Cutoff	3 h, 55 m	Beginning of Rendezvous Preparation	44 days
6.0 Rendezvous Preparations	Beginning of Rendezvous Preparations		Vehicle B Liftoff	
7.0 Vehicle B Earth Ascent	Liftoff	44 days	Parking Orbit Insertion - S-IVB	595.54 sec(1)
7.1 S-IB Thrusting	Liftoff	44 days	S-IB Staging	145.94 sec(1)
7.1.1 Coast		44 d, 141.64 sec		
7.1.2 S-IB Staging		44 d, 145.94 s(1)		4.3 sec
7.2 S-IVB Thrusting	S-IB Staging	44 d, 145.94 s(1)	Parking Orbit Insertion - S-IVB Cutoff	449.60 sec(1)
7.2.1 LES Jettison		44 d, 165.94 s(1)		
8.0 Vehicle B 80-N. Mi. Earth Parking Orbit	Parking Orbit Insertion - S-IVB Cutoff	44 d, 9 m, 55.5 s	TOI - SPS Ignition	4 h, 52 m(3)
8.1 Initial Coast	Parking Orbit Insertion - S-IVB Cutoff		CSM/LEM Separation	

Table 2-5

Mission 2, Saturn IB to Saturn IB Rendezvous Mission Phase (Cont.)

Phase	Initiating Event	Event Time Hrs Min Sec	Concluding Event	Duration ΔT Hrs Min Sec
8.2 Transposition and Docking	CSM/LEM - S-IVB Separation		Hard Dock	40 min
8.3 S-IVB/IU/SLA Staging	Hard Dock		TOI - SPS Ignition	
9.0 Vehicle B Transfer Orbit	TOI SPS Ignition	44 d, 5 h, 02 m	First Terminal Rendezvous Maneuver Ignition	45 m
9.1 TOI Thrusting	TOI SPS Ignition		TOI - SPS Cutoff	11.5 sec
9.2 Transfer Orbit Coast	TOI SPS Cutoff		First Terminal Rendezvous Maneuver Ignition	45 min
9.2.1 MCC No. 1		44 d, 5 h, 24 m		
10.0 Rendezvous of Vehicle A and Vehicle B	First Terminal Rendezvous Maneuver Ignition	44 d, 5 h, 47 m	CSM A Hard Dock	1 h, 27 m (Ref. 26)
10.1 Terminal Rendezvous By Vehicle B	First Terminal Rendezvous Maneuver Ignition		CSM/LEM A Separation	
10.1.1 First Terminal Rendezvous	EOI SPS Ignition		EOI SPS Cutoff	11.24 sec
10.1.2 Second Terminal Rendezvous				

Table 2-5

Mission 2, Saturn IB to Saturn IB Rendezvous Mission Phase (Cont.)

Phase	Initiating Event	Event Time Hrs Min Sec	Concluding Event	Duration ΔT Hrs Min Sec
10.1.3 Third Terminal Rendezvous				
10.2 LEM A Docking Operations	CSM/ LEM A Separation		LEM A-S/C B Hard Dock	
10.2.1 LEM A Separation				
10.2.2 LEM A Begin Soft Dock with S/C B				
10.2.3 LEM A Hard Dock with S/C B				
10.3 CSM A Docking Operations	CSM/LEM A Separation		LEM A-S/C B Hard Dock	
10.3.1 CSM A Begin Soft Dock with S/C B-LEM A				
10.3.2 CSM A Hard Dock with S/C A-LEM A				
11.0 Docked Configuration Coast	CSM A Hard Dock	44 d, 7 h, 14 m	CSM A Separation	1 day(4)
11.1 Crew, Supply and Material Transfer	CSM A Hard Dock		Beginning of CSM A Preparations for Separation	

Table 2-5

Mission 2, Saturn IB to Saturn IB Rendezvous Mission Phase (Cont.)

Phase	Initiating Event	Event Time Hrs Min Sec	Concluding Event	Duration ΔT Hrs Min Sec
11.2 CSM A Preparation for Separation	Beginning of CSM A Preparations for Separation		CSM A Separation	1 hr, 30 m
11.2.1 CSM A Separation				
12.0 CSM A De-Orbit	CSM A Separation	45 d, 7 h, 14 m	Re-Entry	27 min
12.1 SPS Retrothrusting	CSM A Separation		SPS Retrocutoff	12.5 sec
12.2 De-Orbit Coast	SPS Retrocutoff		SM Staging Initiation	17 min
12.3 SM Staging	SM Staging Initiation		Re-Entry	10 min
13.0 CM A Re-Entry	Re-Entry (400, 000 Feet)	45 d, 7 h, 41 m	Drogue Chute Deployment	10 min
14.0 CM A Parachute Deployment	Drogue Chute Deployment	45 d, 7 h, 51 m	Touchdown	7 min
15.0 CM A Recovery and Retrieval	Touchdown	45 d, 7 h, 58 m	Retrieval	
16.0 S/C B Plus LEM A Earth Orbit Coast	CSM A Separation	45 days	Beginning of De-Orbit Preparations S/C B	44 days
17.0 CSM B De-Orbit	Beginning of De-Orbit Preparations S/C B	89 days	Re-Entry	
17.1 LEM A and LEM B Staging	Beginning of De-Orbit Preparations S/C B	89 days	LEM A and B Staging	1 h, 30 m

Table 2-5

Mission 2, Saturn IB to Saturn IB Rendezvous Mission Phase (Cont.)

Phase	Initiating Event	Event Time Hrs Min Sec	Concluding Event	Duration ΔT Hrs Min Sec
17.2 SPS Retrothrusting	LEM A and B Staging	89 d, 1 h, 30m	SPS Retrocutoff	12.5 sec
17.3 De-Orbit Coast	SPS Retrocutoff		SM Staging Initiation	17 min
17.4 SM Staging	SM Staging Initiation		Re-Entry	10 min
18.0 CM B Re-Entry	(400,000 Feet) Re-Entry	89 d, 1 h, 57 m	Drogue Chute Deployment	10 min
19.0 CM B Parachute Deployment	Drogue Chute Deployment	89 d, 2 h, 7 m	Touchdown	7 min
20.0 CM B Recovery and Retrieval	Touchdown	89 d, 2 h, 14 m	Retrieval	

(1) Reference 49, Table 3.
(2) Time established for communications requirements which meets crew task minimum requirements of 2 hr, 52 min.
(3) Variable time required for rendezvous of 0 - 270 min, 2 hrs assumed; crew tasks requirements of 2 hr, 52 min included.
(4) Reference 48 - hard dock time of 1 day established to allow ample transfer time.

2.2.1.2 Phase 1: Preflight

The preflight phase begins with the securing of the launch vehicle to the launch pad and ends with launch vehicle liftoff at T -0.

When the space vehicle is secured and all umbilical interfaces have been verified, the space vehicle subsystems are checked. The GNS is checked and provided with mission information. The liftoff time reference and the real-time data for vehicle performance monitoring is provided. The spacecraft then is fueled.

The LEM and CSM receive cryogenic servicing, cabin, crew, and water supplies. The GNS of both the LEM and CSM are checked out and activated. The CSM is cleaned and sterilized and the launch vehicle propellant is loaded.

The downrange area is cleared, while tracking and communication networks receive final verification. Several hours before launch, the launch vehicle receives its cryogenic load. At T -6 seconds, the S-IB engine is ignited. The vehicle is released and lifts off at T -0.

2.2.1.3 Phase 2: Vehicle A Earth-Ascent

The earth ascent phase (see Appendix C) begins with vehicle liftoff at T -0 and terminates with S-IVB cutoff after parking orbit insertion.

After liftoff, the S-IB rises vertically (pitch = 90 degrees) for 10 seconds, and begins a preprogrammed pitch maneuver that smoothly reduces the pitch angle at S-IB burnout to 26.34 degrees. The time and altitude of the highest axial acceleration are ( ) seconds and ( ) feet. At S-IB burnout (T +141 seconds), the vehicle has a velocity of 7623 fps at an altitude of 199,816 feet. The S-IB is staged prior to S-IVB ignition.

During the ascent, the navigator monitors the S-IB ignition, the caution and warning indicators, and the environmental control system panel. For the duration of the ascent phase, the engineer maintains radio communication with the control center and monitors the critical temperature flow and events displays. The commander reports the S-IB engine cutoff and the S-IB/S-IVB separation, and continually monitors acceleration, attitude, attitude rates, abort lights, and pitch and roll rates.

After verifying S-IVB ignition (T +146 seconds), the commander continues to monitor vehicle performance and at 20 seconds after S-IVB ignition, confirms the LES jettison.

The vehicle pitch angle reduces from 25.6 degrees at S-IVB ignition to zero at the 80-n. mi. earth orbit insertion point. At S-IVB cutoff, T +595 seconds, the vehicle velocity is 25,665 fps, and its altitude is 486,000 feet. Approximately 10 minutes after launch, the commander observes the S-IVB engine cutoff and vehicle insertion into the 80-n. mi. earth parking orbit.

2.2.1.4 Phase 3: Vehicle A Earth Parking Orbit

The earth parking orbit phase begins with S-IVB cutoff and concludes with SPS ignition for transfer orbit insertion.

After S-IVB engine cutoff (at T +595 seconds), the commander and navigator confirm the parking orbit. All three crewmen then adjust their couch restraints and set the controls and displays for parking orbit operations. Using the inflight test system, the navigator and engineer perform a checkout of the spacecraft systems, including guidance and control, environmental control, reaction control, and electrical power. The engineer also continues to maintain communications with MSFN.

The commander next stabilizes the spacecraft attitude so that the navigator can perform the IMU fine alignment and landmark navigational sightings. The S-IVB reaction control system is used to orient the space vehicle for landmark acquisition.

Once he has determined and verified the orbit ephemeris, the commander makes the decision for mission continuation in the normal mode.

The spacecraft then is oriented for transposition and docking, where the sun illuminates the LEM docking tunnel and the sun vector makes an angle of less than 90 degrees with the CSM X-body axis. After the S-IVB SCS is placed in the attitude hold mode, the commander initiates the transposition maneuver, actuating separation of the S-IVB from the CSM.

When the CSM/S-IVB-LEM separation is assured, the SLA is deployed to aid in the stabilization of the S-IVB-LEM and to expose the LEM. The CSM then translates 35 feet forward, performs a 180-degree pitch maneuver, and translates back to the S-IVB-LEM. The CSM performs docking maneuvers to align the CM-LEM interface and to engage the LEM drogue (X-axis docking tunnel). A maximum of 15 minutes is required from separation to the accomplishment of a soft dock.

After soft docking has been achieved, the navigator secures the docking latches and electrical connections between the two modules and pressurizes the LEM. A maximum of 25 minutes is required from soft dock to the securing of the CM latches after performing the hard dock.

Upon completion of hard docking, the commander initiates the S-IVB separation from the CSM/LEM configuration, a sequence visually monitored by the engineer. The commander then translates the CSM/LEM from the jettisoned S-IVB stage. The engineer reports the completion of the maneuver to MCC. A maximum of 5 minutes is required from hard docking to the completion of S-IVB/CSM-LEM separation.

Transposition and docking must be accomplished prior to a mission time of 4.5 hours because the specified lifetime for the Saturn IB IU is 4.5 hours after liftoff.

Following S-IVB staging, the crew prepares for SPS insertion into the transfer orbit. IMU coarse and fine alignments and navigational sightings are made prior to the decision to continue the nominal mission. After the GNS is programmed, the spacecraft is properly oriented and the countdown for TOI is initiated. At a mission time of 3 hours and 10 minutes, the GNS automatically commands SPS ignition. SM RCS ullage ignition occurs 5 seconds prior to SPS ignition.

2.2.1.5 Phase 4: Vehicle A Transfer Orbit

The transfer orbit phase (see Appendix C) begins with the ignition of the SPS for transfer orbit insertion and terminates with SPS thrusting for earth orbit insertion. The nominal mission altitude is 200 n. mi.

During the second 80-n. mi. earth orbit, the SPS is used to transfer the spacecraft into a co-planar Hohmann transfer orbit with an apogee of 200 n. mi.

In preparation for the transfer orbit insertion, the navigator performs both coarse and fine IMU alignments and then calculates the required transfer orbit parameters. The commander maneuvers the spacecraft attitude for the IMU alignment. This onboard trajectory data is compared with MSFN values, and the insertion point is established.

All three crewmen adjust their seat restraints. The commander makes a final check of the SPS and SM RCS for the transfer orbit burn, and initiates the SPS ignition

sequence. All three crewmen monitor the SPS ignition, burn, and cutoff. This maneuver occurs at -84 degrees longitude, 21 degrees latitude.

The navigator confirms the transfer orbit by computing the trajectory parameters and comparing these onboard calculations with the MSFN values. The commander sets the controls for transfer orbit operations, and checks the SPS parameters and propellant status.

During transfer orbit operations, the navigator performs a second IMU alignment in preparation for the second SPS burn into the 200-n. mi. orbit. Also during this period, the engineer checks the electrical power and service propulsion systems. The commander maneuvers the spacecraft for the IMU alignment and maintains communications with the MSFN. The three crewmen then monitor the SPS ignition, burn, and cutoff.

Under GNS guidance, the SPS is used again to insert the spacecraft into a circular orbit at the nominal 200-n. mi. mission altitude. This thrusting maneuver occurs at 84 degrees longitude, -21 degrees latitude, and T +3 hours and 55 minutes after launch. The termination of EOI SPS thrusting ends the transfer orbit coast phase.

2.2.1.6 Phase 5: Vehicle A Earth Orbit Coast

The 200-n. mi. earth orbit phase (see Appendix C) begins with EOI SPS cutoff and terminates on the day when rendezvous preparations are begun.

Upon completion of the SPS orbit insertion burn, the CSM subsystem checks are accomplished. MSFN verification of the ephemeris is obtained, and the CSM subsystems are prepared for the orbital coast.

The navigator confirms orbit injection using the spacecraft displays, the guidance computer, and data from MSFN. He secures the controls and displays and assists the commander in a CSM systems status check. While the engineer continues to record PCM telemetry and maintain ground communications, the navigator proceeds with the fine alignment and the landmark navigation sightings. During these operations, the commander maneuvers the vehicle attitude for landmark acquisition. The navigator next compares his orbit information to ground-computed parameters to insure the accuracy of the onboard calculations.

The engineer prepares to transfer to the LEM for earth orbit preparations. The commander checks the LEM pressurization from the CM. Then, the engineer equalizes the CM and interlock pressure, removes and stows the CM pressure and thermal hatches, and disconnects and stows the probe and drogue. Thermal cycling is accomplished when necessary. After activating and checking his portable life support system, the engineer transfers from the CM to the LEM by proceeding through the tunnel. Upon completion of an inspection of the LEM cabin, he closes the hatch and initiates communications with MSFN. After a brief checkout period, experiments that require crew activation or participation are set up.

During each day of the orbit phase, the crew's primary activities include scientific experiments, systems status checks, and navigational sightings. Personal functions are performed in the time remaining. Detailed crew task schedules for day 1 through day 14 are shown in Figure 2-10.

2.2.1.7 Phase 6: Rendezvous Preparations

Rendezvous preparations (see Appendix C) begin on the forty-fourth mission day of Vehicle A. The phase terminates with Vehicle B liftoff toward the rendezvous launch window.

During this period, MSFN continuously monitors the position of Vehicle A in the mission orbit, and periodically updates the launch computer. MCC integrates the trajectory of Vehicle A continuously from data supplied by MSFN.

Nominally, on the forty-fourth day of Vehicle A orbit operations, Crew B enters Vehicle B preparing for the launch and subsequent rendezvous and docking with Vehicle A. The commander arms the LES, and the crew members check out their pressure suits. The navigator and engineer then monitor vehicle systems, and the mission controller considers the decision to launch. The decision will be based on the relative position of Vehicle A and the countdown time of Vehicle B. Final countdown is entered and launch is initiated.

While Spacecraft B is being checked out and launched, the crew of A is transferring material between the LEM and the CSM. Subsystems necessary for rendezvous operations are checked. The crew then awaits the launch of Spacecraft B.

Spacecraft B is launched only when an optimum rendezvous launch window is available because of weight limitation. With S-IB liftoff, the rendezvous preparations phase is terminated.

2.2.1.8 Phases 7, 8, and 9: Vehicle B Earth-Ascent, Earth Parking Orbit, and Transfer Orbit

The earth ascent phase (see Appendix C) begins with Vehicle B liftoff, on Vehicle A's forty-fourth mission day, and terminates with S-IVB cutoff after parking orbit insertion. Phases 7, 8, and 9 for Vehicle B are the same as Phases 2 through 4 for Vehicle A (see paragraphs 2.2.1.3 through 2.2.1.5), with exception of the following paragraphs.

During the final phase of the transfer orbit coast, the crew of Vehicle B prepare for a midcourse guidance correction. The commander and MSFN decide if a correction is necessary. Increases in parking orbit coast times will be made if the launch is late (see Part II, Section 5).

The engineer maintains communications with MSFN, holds Spacecraft A on radar, and periodically updates the guidance computer. Based on this tracking data, the navigator computes a revised transfer orbit, which is verified by MSFN. The commander maintains Vehicle B attitude, and the crews of both vehicles check the mutual ranging parameters, range, and range-rate data. The engineer continues to monitor the CSM B systems status, and the commander orients Vehicle B for the SM RCS firing sequence. At the termination of the commander's countdown, the navigator monitors SM RCS ignition, burn, and shutdown. After confirming the effect of the midcourse trajectory correction, the navigator performs an IMU alignment, while the commander stabilizes Vehicle B attitude. The commander maintains communications with Vehicle A and MSFN. As a preparatory step to the SPS burn into the 200-n. mi. orbit, the engineer checks out the electrical power and SPS.

2.2.1.9 Phase 10: Rendezvous of Vehicles A and B

Phase 10 begins, at a mission time of T +44 days, 5 hours, and 47 minutes, with Vehicle B's SPS ignition for the initial terminal rendezvous maneuver, and ends with CSM A hard docked to the Spacecraft B-LEM A configuration.

At the initiation of rendezvous, the LOS range is ( ) n. mi., and the LOS range rate is ( ) fps.

The three crewmen of Vehicle B monitor the SPS ignition, burn, and cutoff. For the duration of the rendezvous phase, radar continues to supply tracking data, and the VHF link provides communication between the two spacecraft. Following the navigator's confirmation of the 200-n. mi. orbit, the engineer initiates control of the line-of-sight indicators and correlates the range-rate parameters between Vehicles A and B. The navigator checks the mutual radar ranging parameters and verifies the trajectory of Vehicles B and A with MSFN. With this data, the onboard computers calculate the total velocity increment necessary to achieve final rendezvous.

The SM RCS is thrusted for ( ) seconds, reducing the LOS range rate of ( ) fps as scheduled. The vehicle then coasts for ( ) seconds until an LOS range of ( ) n. mi. is attained. The second thrusting commences and lasts for ( ) seconds; thus thrusting reduces the closing rate to ( ) fps. Another coast period of ( ) seconds precedes the final thrust period of ( ) seconds at a LOS range of ( ) n. mi. The closing rate is reduced to ( ) fps. (Detailed parametric data or rendezvous is not available.)

The commander maintains the proper attitude of Vehicle B, and controls the relative velocity along the line-of-sight by monitoring the automatic SM RCS firings and the forward velocity indicator. Additional radar data, plus visual sightings of target Vehicle A, enable the commander to brake to a closing velocity of approximately 5 fps. At a range of 500 feet, the crews of both vehicles prepare for final rendezvous and docking maneuvers.

The final rendezvous and docking phase requires the transfer of two crew members from the CSM A to the LEM A, the separation of CSM A and LEM A, the docking of LEM A with LEM B, and the docking of CSM A with the resulting LEM A, LEM/CSM B configuration. Figure 2-1 shows in graphic form the steps required to rendezvous the two spacecraft.

After the navigators of Vehicles A and B have checked and verified the results of the final rendezvous thrust maneuver, the Vehicle A commander and engineer transfer to the LEM A and prepare for separation. The A crew members, including the navigator at the control panel in the CM, fasten their couch restraints. The A commander initiates LEM/CSM separation by igniting the shaped charges to sever the LEM/CSM interface. Using the LEM A RCS, the commander translates the LEM A from the CSM A, performs the necessary attitude and translation maneuvers for a soft docking with the LEM A, and obtains event and status verification. After soft docking is

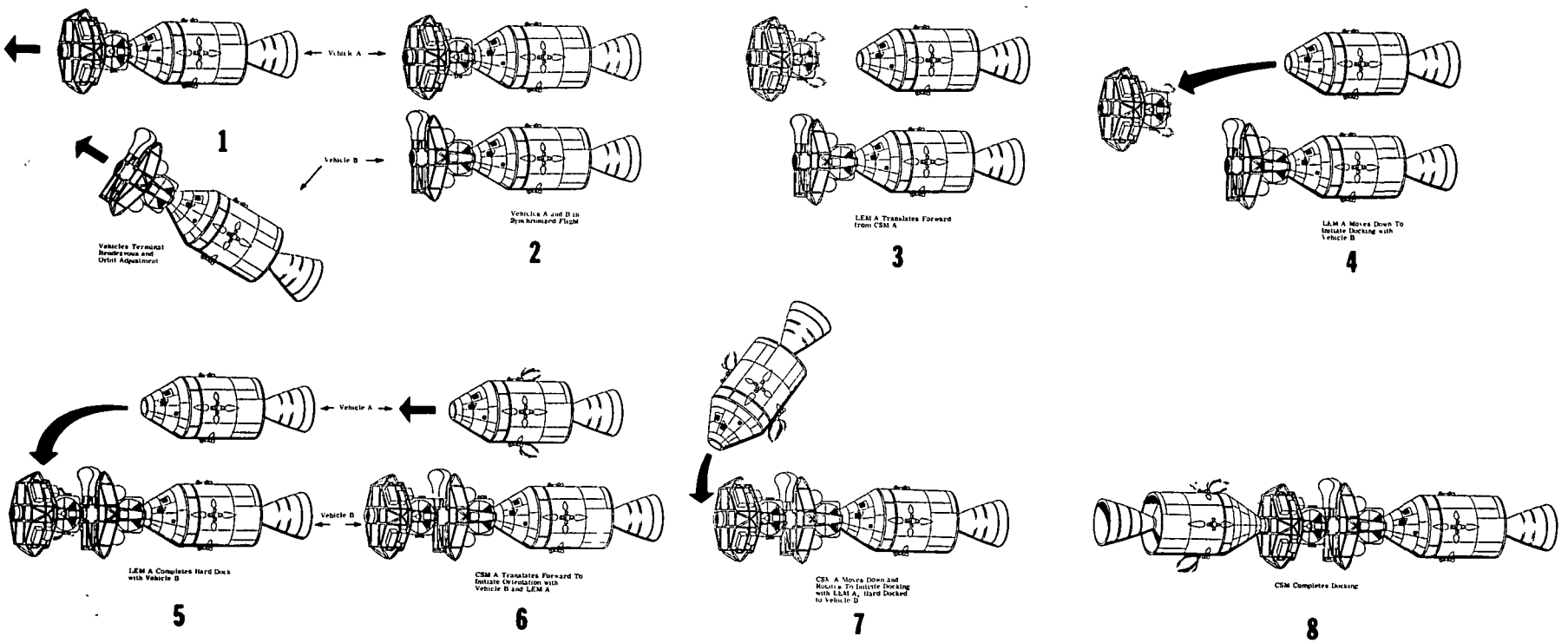


Figure 2-1. Terminal Rendezvous and Docking Sequence

achieved, the B navigator secures the docking latches and electrical connections between the two LEM's and equalizes the cabin pressures.

The A navigator then performs a similar rendezvous maneuver to dock CSM A with the docking ring on the descent stage of LEM A. In both docking maneuvers, the A modules are the active vehicles, the CSM/LEM B configuration remains passive. Both crews now prepare for personnel and equipment transfer.

After engagement, hard-dock proceedings are begun. With hard dock completed, the two spacecraft enter the docked configuration coast phase.

2.2.1.10 Phase 11: Docked Configuration Coast

With the completion of CSM A hard dock to Spacecraft B plus LEM A, the two spacecraft coast together in earth orbit until CSM A separates for de-orbiting (see Appendix C). With the A and B spacecraft mated at the interface between the two LEM's, the crews transfer cargo through the docking tunnels. This transfer includes the removal of active experiments from the CSM B and the replacement of expended LEM A materiel from Vehicle B supplies. Upon completion of this cargo transfer, the A commander and B engineer join the A navigator in the original CSM. The A engineer remains in the CSM-LEM B/LEM A configuration with the replacement commander and navigator. Both crews prepare for the separation and re-entry of CSM A 24 hours after docking.

After all crew members have fastened their couch restraints, the A commander initiates CSM A separation by igniting the shaped charges to sever the LEM A/CSM A interface. He then translates the CSM A from LEM A, using the SM RCS. The commander maneuvers Spacecraft A to the proper attitude, while the navigator performs a navigation sighting. After completing the IMU fine alignment and a landmark navigational sighting, the navigator computes the onboard trajectory. The MSFN supplies orbital parameters for comparison with the navigator's trajectory calculations. Crew A then checks all CSM systems in preparation for the third SPS burn into the de-orbit coast phase.

CSM A separation occurs at mission time T +45 days, 6 hours, and 14 minutes. Spacecraft B now enters its earth orbit coast phase, while CSM A begins de-orbit preparations.

2.2.1.11 Phase 12: CSM A De-Orbit

The de-orbit phase (see Appendix C) begins when the crew initiates de-orbit preparations and terminates with the CM entry into the earth's atmosphere at 400,000 feet.

Preparations for CSM A separation are begun several hours prior to separation. The crew installs and secures the hatches, orients the spacecraft for the separation maneuver, initiates the CM-LEM separation devices, and ignites the SM RCS to translate the SM away from the Spacecraft B/LEM A configuration. The CSM is separated at a mission time of T +45 days, 6 hours, and 14 minutes at a geocentric location of 200-n. mi. altitude, -120 degrees longitude, and 28 degrees latitude. MSFN tracking is available during this maneuver. CSM A separation nominally occurs prior to the last orbit.

After CSM/LEM separation, the crew begins de-orbit preparations. The de-orbit maneuver parameters are determined with MSFN assistance and verification. Following verification of the navigator's de-orbit parameters by MSFN, the three crewmen don their pressure suits and adjust their restraints. The CSM GNS is programmed, and the CSM is oriented before the crew initiates the countdown for SPS retroignition. At the termination of the countdown, the CSM GNS automatically commands SM RCS ullage ignition and SPS ignition at the proper time.

An SPS retrofiring of 12 seconds inserts the CSM into a descending transfer orbit having a vacuum perigee of zero n. mi. The SPS thrust initiation begins at a mission time of 45 days 7 hours 14 minutes and at 47.0 degrees longitude and 28.5 degrees latitude.

The commander checks the insertion parameters and the CSM systems status, and communicates his findings to the MSFN. Next, he prepares for SM jettison by orienting the CSM to the proper attitude and activating the CM/SM separation sequence. During this period, the engineer switches all systems from SM to CM, and retracts the antenna and optics.

After a final IMU alignment, the CSM is properly oriented and the SM staged. SM staging occurs 10 minutes prior to re-entry. At the time of SM staging, the spacecraft is 190 n. mi. above the earth's surface.

All three crewmen monitor the CM/SM separation. After SM separation, the navigator and engineer perform a post-separation status check of the CM RCS. The commander then maneuvers the CM to the MSFN confirmed entry attitude. Following a check by

the commander of the entry monitor flight indicator, the crew members adjust restraints, check suit umbilicals, and close faceplates.

The de-orbit phase ends when the CM re-enters the earth's atmosphere at a nominal altitude of 400,000 feet. Re-entry occurs at a geocentric longitude and latitude of -145 and 8 degrees, respectively.

2.2.1.12 Phase 13: CM A Re-Entry

The re-entry phase begins when the CM enters the earth's atmosphere and ends at the time of drogue chute deployment, which occurs at an altitude of 24,000 feet.

For the preselected landing site target of -158 degrees and 28 degrees, the range-to-go at the time of entry is ( ) n. mi. The entry velocity, flight path angle, inclination, and azimuth are 25016 fps, ( ) degrees, ( ) degrees, and ( ) degrees, respectively.*

The initial part of the re-entry phase is a full-lift flight from 400,000 feet to a point where the rate of descent is reduced to 700 fps. At this point, trajectory range control is initiated. During trajectory range control, the steering commands are based on a reference trajectory, selected where the guided trajectory range from the control initiation points to touchdown matches the range requirements to reach the target. The final part of the re-entry phase consists of an equilibrium glide to an altitude of 24,000 feet, the altitude at which drogue chute deployment occurs.

During re-entry the commander continues to monitor the CM entry attitude and the glide maneuver. The navigator starts the elapse event time clock upon 0.05g indication. The engineer maintains communications throughout this re-entry phase with the MSFN recovery forces. At 100,000 feet, the commander arms the earth landing system and monitors the descent to 25,000 feet.

The forward heat shield is jettisoned at an altitude of 25,000 feet, 3 seconds prior to drogue chute deployment, which marks the beginning of the parachute descent phase.

*Detailed re-entry trajectory parameters from an altitude of 200 n. mi. with a vacuum perigee equal to zero are not available.

2.2.1.13 Phase 14: CM A Parachute Descent

The parachute descent phase begins at an altitude of 24,000 feet, at the time of drogue chute deployment, and ends approximately 7 minutes later with CM earth landing.

The parachute descent phase begins at a mission time of T +45 days, 7 hours, and 51 minutes, when the CM is at an altitude of 24,000 feet and approximately over the earth landing site location. The CM relative velocity and flight path angle at the point of drogue chute deployment is ( ) fps and ( ) degrees. The landing site is at a geocentric longitude and latitude of -156 and 28 degrees, respectively.

The navigator monitors the forward heat shield deployment at 25,000 feet and the drogue chute deployment 1.5 seconds later. If parachute deployment does not occur automatically, he deploys the chute manually.

Fifty-eight seconds after the beginning of the phase, the drogue chutes are jettisoned and main chutes are deployed at an altitude of 11,000 feet. Earth landing occurs at 420 seconds after the deployment of the drogue chutes. At CM landing, the relative velocity and flight path angle are ( ) fps and -90 degrees, respectively. After the navigator activates the VHF systems, the three crewmen shut off all power switches not required for landing. The remaining propellant in the CM RCS is then dumped by the navigator. The commander continuously monitors the altitude and descent velocity during this phase, communicating the status of the descent to the MSFN.

Recovery communications (VHF) are activated and a final systems check is performed during parachute descent, starting 4 minutes and 49 seconds prior to touchdown.

2.2.1.14 Phase 15: CM A Recovery and Retrieval

The recovery and retrieval phase begins with CM A touchdown at a mission time of T +45 days, 7 hours, and 58 minutes, and continues until both the CM and crew are onboard a recovery ship.

Immediately after touchdown, each crew member releases his seat restraints, and assesses his physical condition and the CM structural condition. While the commander activates the post-landing recovery aids, the engineer checks the EPS and ECS for post-landing operations. After obtaining survival equipment from the storage area, Crew A egresses to await recovery.

2.2.1.15 Phase 16: Spacecraft B-Plus-LEM A Earth Orbit Coast

The earth orbit coast period for the Spacecraft B/LEM A configuration begins with the separation of CSM A from the docked configuration of the two spacecraft (see Appendix C). Termination of this phase occurs at the time of de-orbit preparation. The mission duration and altitude are nominally 45 days and 200 n. mi.

Experiments that require crew activation or participation are initiated, or set up, according to the approved experiment plan. The crew work/rest cycle is started. The three crew members remaining in the CSM-LEM B/LEM A for the second phase of the 90-day orbit observe the same work/rest cycle as outlined for Crew A in paragraph 2.2.1.6. Periodic status checks of the CSM and LEM subsystems, and periodic alignments of the IMU are accomplished on a scheduled basis. Thermal cycling will be accomplished as required during the 45-day mission.

On the last mission day, during the final orbits, preparations for de-orbit are begun, and the crew and materiel are transferred from the LEM's to CSM B. Phase 16 ends with these de-orbit preparations.

2.2.1.16 Phase 17: CSM B De-Orbit

The de-orbit phase begins when the crew initiates de-orbit preparations and terminates with the CM entry into the earth atmosphere at 400,000 feet.

Preparations for LEM A/LEM B staging are begun at a mission time of 89 days. The crew installs and secures the hatches, orients the spacecraft for LEM staging, initiates the CM-LEM separation devices, and ignites the SM RCS to translate the CSM away from the LEM A/LEM B configuration. The LEM's are staged at a mission time of 89 days 0 hours 30 minutes at a geocentric location of 200-n. mi. altitude, -120 degrees longitude, and 28 degrees latitude. MSFN tracking is available during LEM staging. LEM staging nominally occurs immediately prior to the last orbit.

After CSM-LEM separation, the crew begins de-orbit preparations. The de-orbit maneuver parameters are determined with MSFN assistance and verification. The CSM GNS is programmed, and the CSM is oriented before the crew initiates the countdown for SPS retroignition. At the termination of the countdown, the CSM GNS automatically commands the SM RCS ullage ignition and SPS ignition at the proper times.

An SPS retrofiring of 12 seconds inserts the CMS into a descending transfer orbit having a vacuum perigee of zero n. mi. The SPS thrust initiation begins at a mission time of 89 days 1 hour 30 minutes and at 47 degrees longitude and -28.5 degrees latitude.

With the termination of SPS retrothrusting, the crew accomplishes the final CSM subsystems status checks, coordinates with MSFN on re-entry, and begins preparations for CM-SM separation. After a final IMU alignment, the CSM is properly oriented and the SM staged. SM staging occurs 10 minutes prior to re-entry. At the time of SM staging, the spacecraft is 190 miles above the earth's surface, at 115 degrees longitude and -5 degrees latitude.

Following SM jettison, a final pre-entry check is conducted which includes an RCS and GNS entry check, and entry monitor systems check, a crew equipment systems check, and a CM orientation for re-entry. The de-orbit phase ends when the CM re-enters the earth's atmosphere (at a nominal altitude of 400,000 feet) at a geocentric longitude and latitude of -145 degrees and 8 degrees, respectively.

2.2.1.17 Phases 18, 19, and 20: CM B Re-Entry, Parachute Descent, and Recovery and Retrieval

Re-entry for CM B is the same as for CM A (see paragraphs 2.2.1.12 through 2.2.1.14).

2.2.2 CONFIGURATION

2.2.2.1 General

The command modules used in Mission 2 are NAA Block II Mod. D configurations. The service module is NAA Block II Mod. D configuration, except for (a) RCS fuel loadings, that are based on Grumman AES studies (see 32 1-7 1-8); and (b) the SPS retrofuel loads that were computed for this study (see Part II, Section 6). Table 2-6 provides the changes to the Block II CSM systems made by the Mod. D configuration.

2.2.2.2 Flight A Configuration

The Flight A (218) configuration carries equipment for experiments dealing with the biosciences and physical sciences. Figure 2-2 shows the location of experiment equipment. Conceptual sketches are available for several of the pieces of equipment used in the mission and are shown in Figure 2-3.

Table 2-6

Block II CSM Configuration Changes

CSM System	Added	Removed
Communications and Data	Wiring and Wiring Interface	High-Gain and Rendezvous Transponder
Environment Control	Blower and Duct Spare Compressor Cabin Lines	
	100 psi O_2 Line Modified Compressor Cabin Fans	
Electrical Power	Wiring Interface	
SM Reaction Control	2 Sets of LEM Tanks per Quad	
Stabilization and Control	Horizon Sensors Redundant Units	

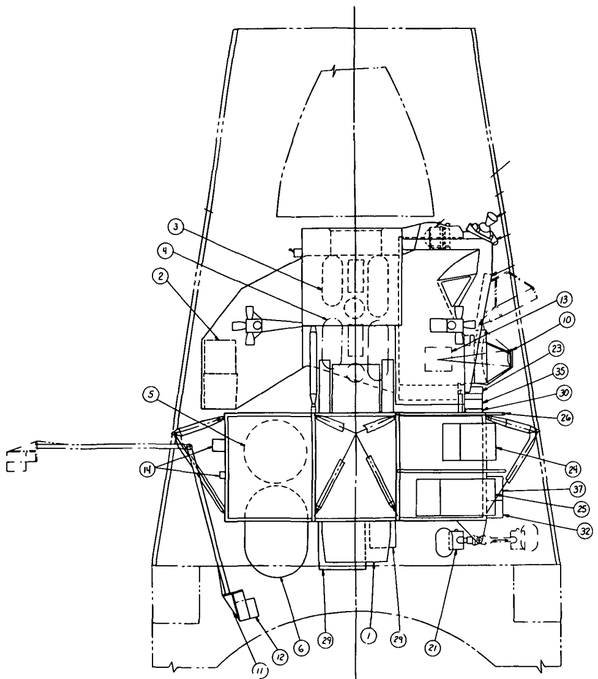
The cosmic ray emission experiment (0606) is mounted on the aft boom during extravehicular activity (EVA), and requires earth orientation. The solar concentrator of the crystallization studies (0605) requires orientation toward the sun. Attitude is held within a deadband of ±5 degrees, except during experiment periods having specific pointing and stability requirements. Pointing and stability requirements are accommodated by the CSM within the RCS propellant capacities.

The bulk of the equipment for EVA's (1501, 2, 3, 4, 5) are located externally in the forward portion of the LEM descent stage. The solar concentrator (0605) is rotated away to permit ingress and egress through the forward hatch. The transfer experiments (1504) are conducted prior to docking. A motor reel assembly, located on the descent stage below the forward hatch, deploys a line for the transfer of materiel between the two spacecraft.

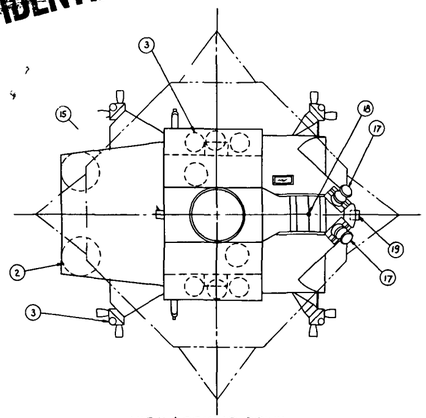
Structural modifications to the LEM ascent stage are required to accommodate the solar concentrator quartz window in the forward hatch. All other experiment equipment installations also involve additions to existing LEM ascent and descent stage structures.

KEY

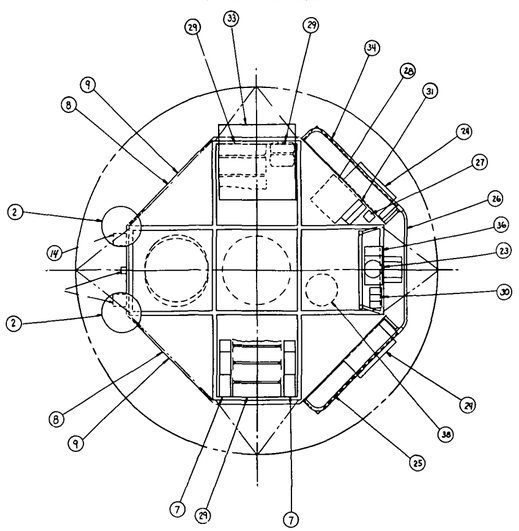
No.	Experiment	Name
1		Docking Ring
2		FCA
3		RCS
4		Supplementary RCS (Flt 7 D only)
5		O_2
6		H_2
7		LiOH
8		Equipt Radiator
9		FCA Radiator
10	605	Solar Concentrator
11	606	Boom for Cosmic Ray Emulsion Exper
12	606	Magnetometer
13	605	Crystallization Studies Test Chamber during exper
14	0501	Radiation Monitor Equipt - Electron Spectrometer (2) Proton Alpha Spectrometer (1)
21	1501,2,3,4,5	Cameras (TV & Movie)
22	1501	ECS Station
23	1501	Safety Line
24	1501	AMU
25		Hard Space Suit
26		Rail & Platform
27		AMU Resupply Station
28	1501	AMU Fuel Supply
29	1502	Cargo Net
30	1502	Tethers (3)
31	1502	AMU Spare Batteries
32	1503	Personnel Transfer Equipt
33	1503	Logistics Shapes Storage
34	1503	Tools for Rescue & Pickup of Objects
35	1504	Motor Reel
36	1504	Rope Gun
37	1505	EVA Tool Kit
38		GOX



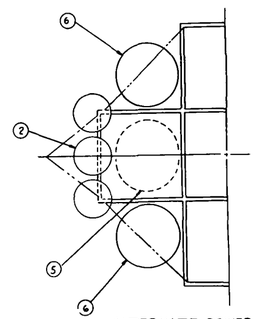
LAUNCH CONFIGURATION



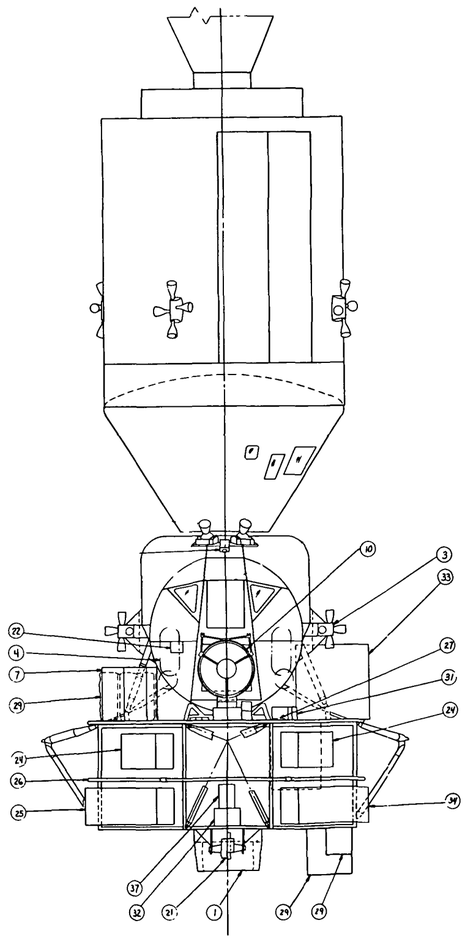
VIEW LOOKING DOWN AT ASCENT STAGE



VIEW LOOKING DOWN AT DESCENT STAGE



ALTERNATE POWER PACK CONFIGURATION



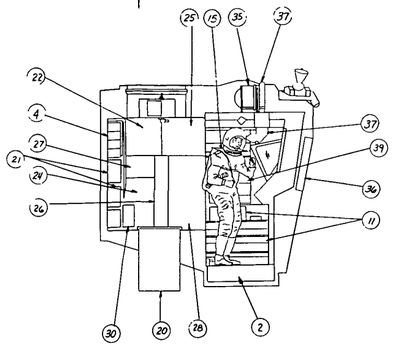
ORBITING CONFIGURATION

0 20 40 60 80 100

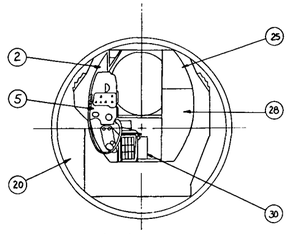
Figure 2-2. Spacecraft Experiment Configuration, Mission 2, Launch A (Sheet 1 of 2) (32 pages 1-7.29)

KEY

No.	Experiment	Name
1		Air Duct
2		Food
3		GOX Tank
4		LioH
5		ECS
6		O_2
7		FCA
8		H_2
9		Attitude Controller Assy
10		Translation Controller Assy
11		Recorders
12		S & C, Power, N&G, ECS, RCS, Comm Panel
13		RCS
14		Docking Ring
15	501	Radiation Environment Monitor Panel
16	501	Radiation Environment Monitor Equipt
17	0100 & 0200	Biomed Incubator
18	0100 & 0200	Biomed Refrigerator
19	0100 & 0200	Biomed Panel
20	0100 & 0200	Biomed Storage
21		CDU, LGC, PSA & Signal Conditioner
22	0601	Capillarity Studies Exper.
23	0602A	Kinetics & Dynamics of Vapor /Gas Exper.
24	0602B	Liquid Drop Dynamics Study Exper.
25	0603A	Pool Boiling in Long Term zero-g Environment Exper.
26	0603B	Nucleate Condensation of Fluids in zero- Exper.
27	604	Density Gradient of a Fluid at near-critical state Exper.
28	605	Crystallization Studies Exper. (Stowed)
29	605	Solar Concentrator
30	606	Cosmic Ray Emulsion Exper. (Stowed)
31	606	Cosmic Ray Emulsion Exper. (Exposed)
32	606	Magnetometer



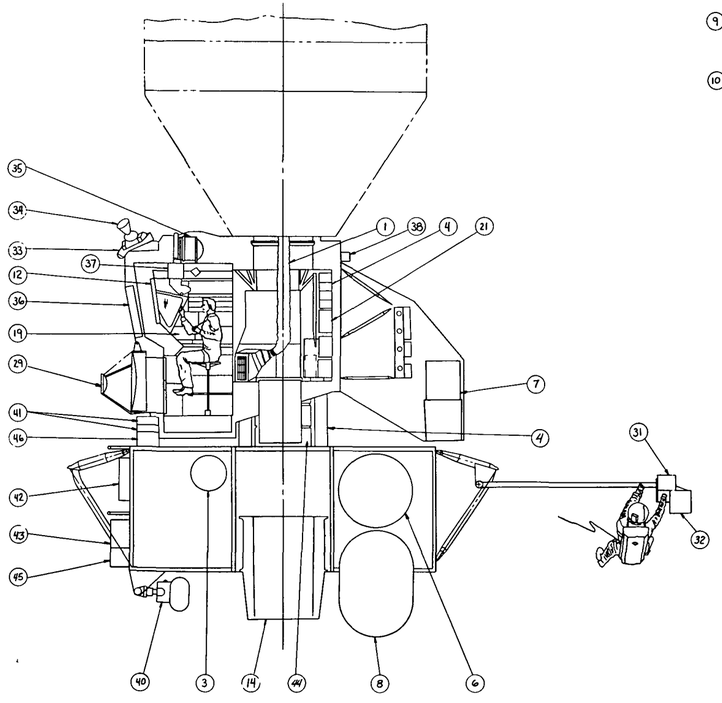
VIEW LOOKING OUTBOARD
LEFT HAND SIDE



VIEW LOOKING AFT

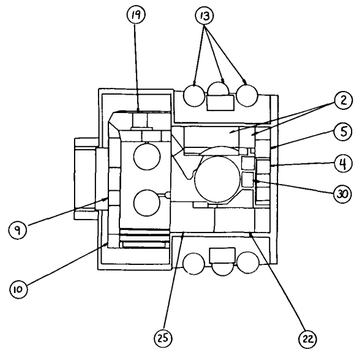
35	on	IMU & Electronics
36	Flt.	Doppler, Altimeter
37	219	Telescope
38	1407	Sun Sensor
39	1500	Extravehicular Operations Panel
40	1500	Camera
41	1501,2,3,4,5	Safety Lines
42	1501,2,3,4,5	AMU
43	1505	EVA Tool Kit
44	1502	Cargo Net
45	1503	Personnel Transfer
46	1504	Motor Reel

CONFIDENTIAL

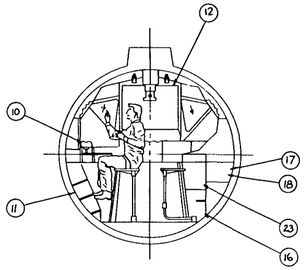


VIEW LOOKING OUTBOARD
RIGHT HAND SIDE

0 20 40 60 80 100



PLAN VIEW



VIEW LOOKING FORWARD

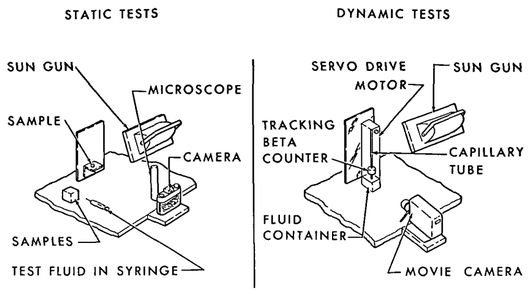
CONFIDENTIAL

Figure 2-2. Spacecraft Experiment Configuration, Mission 2, Launch A
(Sheet 2 of 2) (32 pages 1-7.30)

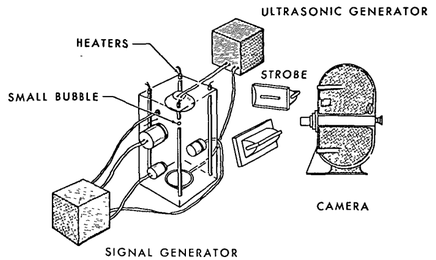
CONFIDENTIAL

CONFIDENTIAL

CAPILLARITY STUDIES - 0601



DYNAMICS OF VAPOR/GAS BUBBLES - 0602A



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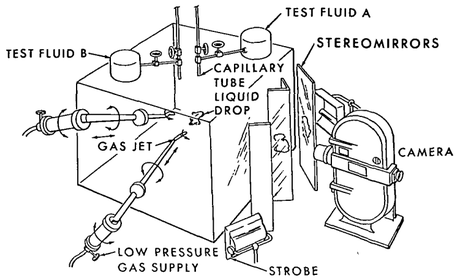

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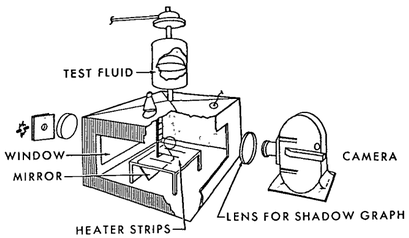

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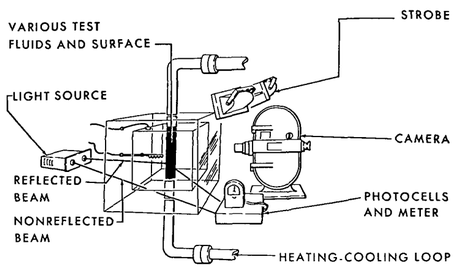
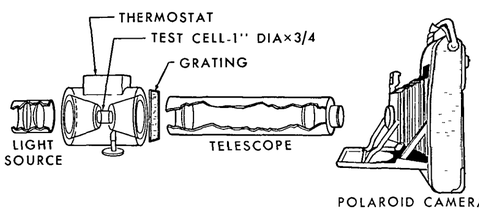
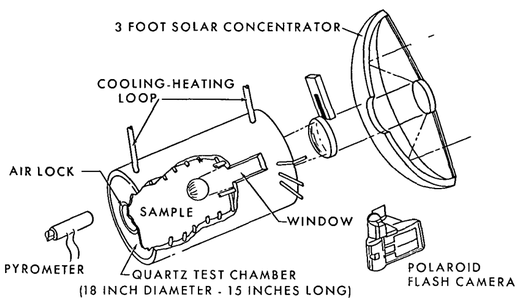


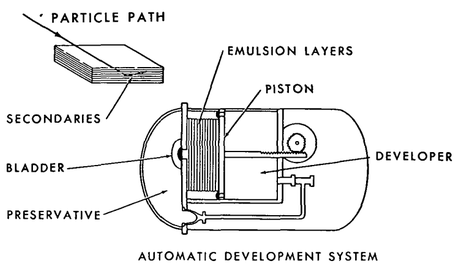
FLUID AT A NEAR CRITICAL STATE - 0604



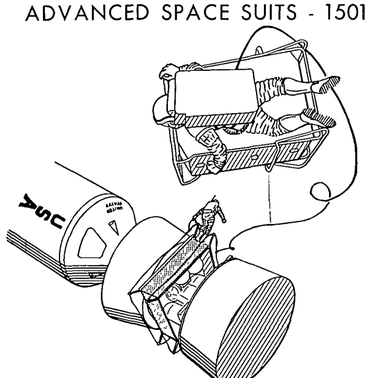
CRYSTALLIZATION STUDIES - 0605



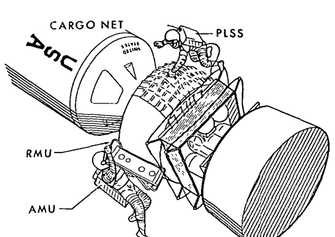
COSMIC RAY EMULSION - 0606



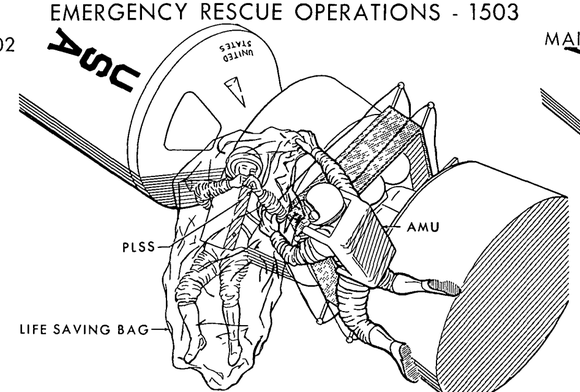
ADVANCED SPACE SUITS - 1501



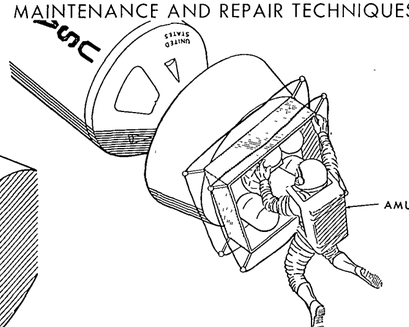
MANNED LOCOMOTION & MANEUVERING - 1502



EMERGENCY RESCUE OPERATIONS - 1503



MAINTENANCE AND REPAIR TECHNIQUES - 1505



ON-BOARD CENTRIFUGE - 0302

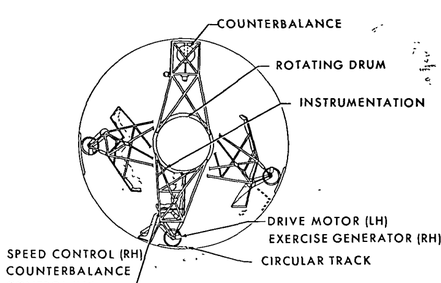


Figure 2-3. Conceptual Sketches for Mission 2 Equipment (See 11, 13)

2-39-1

2-39-2

Adequate internal volume is available to accommodate experiments. Experiment displays and controls, including the LEM left-hand flight station, are located in the forward cabin, or are accessible from forward cabin crew positions. Food is stored below the floor in the forward compartment and in the equipment tunnel. Lithium hydroxide canisters are stored internally, mounted on the aft bulkhead of the laboratory, and replaced from external storage during programmed EVA's.

This flight utilizes an Apollo CSM fuel cell electrical system, a LEM ECS to provide independent life support during rendezvous, and selected components to provide data transmission, status, and appropriate displays.

VHF communications are carried in the laboratory to provide voice and data links with the CM during rendezvous.

2.2.2.3 Flight B Configuration

The second flight configuration carries resupply and supplementary experiments dealing with bioscience and EVA, to be evaluated after rendezvous with the previous flight. Figure 2-4 shows the location of the experiment equipment. This flight carries the 0302 onboard centrifuge. Attitude is held within a deadband of ±5 degrees, except during experiment periods having specific pointing and stability requirements.

The laboratory is a standard LEM ascent stage modified for living organism experiments. The living organism experiments (0401, 2, 3, 4, 5) are located in the forward cabin and equipment tunnel of the ascent stage. The incubators and refrigerator associated with the spaceborne microorganisms experiment (0406) are located in the forward cabin of the ascent stage, while the cartridges are stowed on the right-hand side of the descent stage. All animal containers are capable of pressurization and contain an ECS loop.

All experiment equipment installations are additions to existing LEM ascent stage structure. A low profile descent stage (30-inch depth) is used, which contains a docking tunnel and supports the onboard centrifuge (0302). The tunnel permits passage through the existing ascent stage engine floor cutout in coveralls. Pressure suits and backpacks may be passed down as required. The centrifuge is operated in a shirtsleeve environment.

Adequate internal volume is available to accommodate experiments. All primary experiment displays and controls for this flight are located in the forward cabin of laboratory A. A freezer for the living organism experiments (0401, 2, 3, 4, 5) is located below the floor in the forward cabin. All resupply lithium hydroxide canisters are stored externally in the descent stage. Resupply food is stored in the resupply laboratory cabin and restocked into the previous flight laboratory once docking is achieved.

In addition to the standard CSM/LAB interface, the B flight requires additional interface connections between this laboratory and the originally resupplied laboratory. The interface wiring is the same as that required for the CSM/LAB interface.

This flight utilizes an Apollo CSM fuel cell electrical generation system, and a LEM ECS to provide atmosphere revitalization to the animal tanks, and recirculate the atmosphere of both the CM and laboratory A. It is anticipated that open hatch operation can be made available for laboratory B if desired.

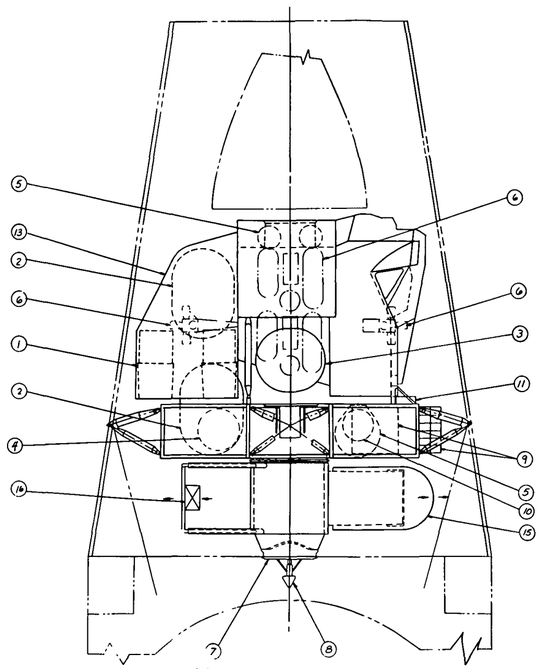
The power supply carried on this flight is sized to provide more than adequate housekeeping power.

2.2.2.4 Configuration History

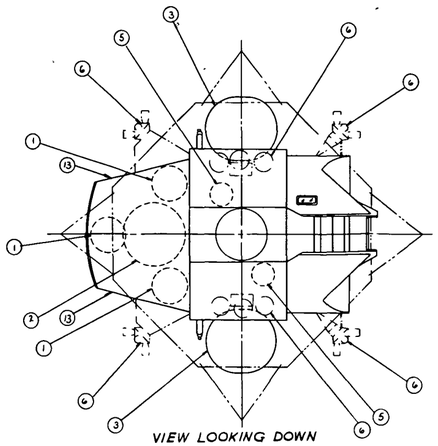
The configuration history for the Saturn IB to Saturn IB Rendezvous mission divides the mission profile into five major operations areas as shown in Figure 2-5. The launch vehicle and spacecraft configuration are depicted for each of the major areas. Mission times from the trajectory analysis and mission phasing charts are shown, relating time to changes in configuration.

KEY

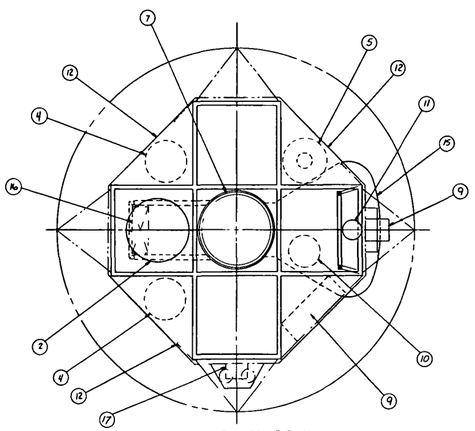
No.	Experiment	Name
1		Fuel Cell Assy
2		H_2 (Block III)
3		O_2 (Block III)
4		O_2 (Block II)
5		Water Tanks
6		RCS (on 8 'D Only)
7		Tunnel & Docking Ring
8		Docking Probe
9		LiOH
10		GOX
11		Safety Line
12		Equipment Radiator
13		Fuel Cell Radiator
14		Flight 7
15	0302	On-Board Centrifuge
16	0302	Centrifuge Counterweight
17	0406	Spaceborne Micro-organism Cartridges (Stowed)
18	0406	Spaceborne Micro-organism Cartridges (Exposed)



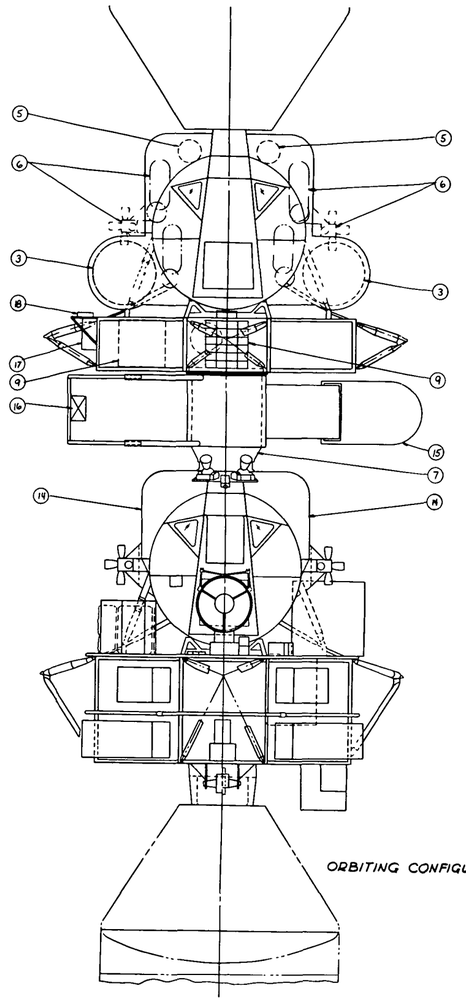
LAUNCH CONFIGURATION



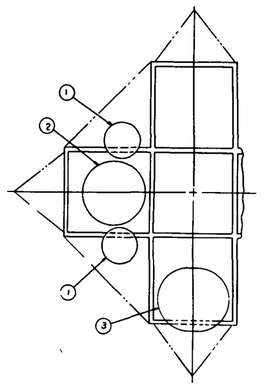
VIEW LOOKING DOWN AT ASCENT STAGE



VIEW LOOKING DOWN AT DESCENT STAGE



ORBITING CONFIGURATION



ALTERNATE POWER PACK CONFIGURATION

Figure 2-4. Spacecraft Experiment Configuration, Mission 2, Launch B (Sheet 1 of 2) (32 pages 1-8.31)

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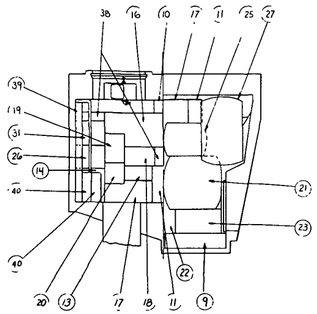
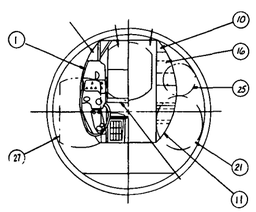
KEY

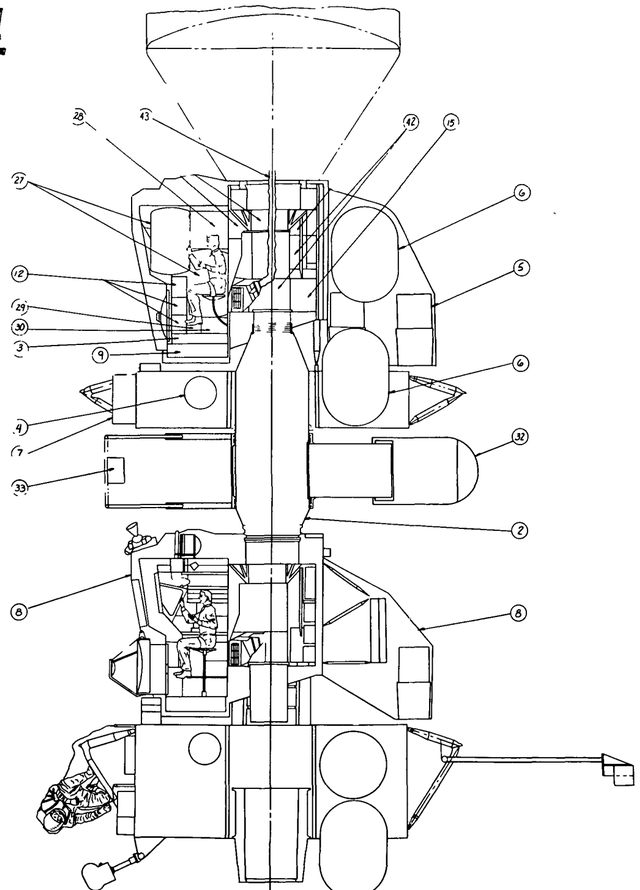
No.	Experiment	Name
1		ECS
2		Tunnel & Docking Ring
3		Film Storage
4		GOX
5		Fuel Cell Assy
6		H_2
7		LiOH
8		Flight 7 Lab
9	401,2,3,4,5	Freezer
10	401	Lyophilization
11	401,403,404	Chemical & Equipment Storage
12	401,402,406	Incubators
13	401,402,406	Microscope
14	401,402	Refrigerator (-15°c)
15	402	Refrigerator (-20°c)
16	402	Culture Flasks
17	402,403,404	Supply Cabinets (Includes 35mm Camera)
18	402	Scintillation Counter
19	402	Particle Counter
20	402	Centrifuge
21	403	Chimpanzee Container (Pressurized)
22	403	Waste Management (Chimpanzee)
23	403	Food Management System
24	403	Instrumentation for Chimpanzee Panel
25	404	Planaria & Newts Chamber (Pressurized)
26	404	Refrigerator (-5°c)
27	405	Rats Cages (Pressurized)
28	405	Food Storage
29	405	Waste Storage
30	405	Quick Freezer
31	406	Refrigerator (+5°c)
32	302	On-Board Centrifuge
33	302	Centrifuge Counterweight



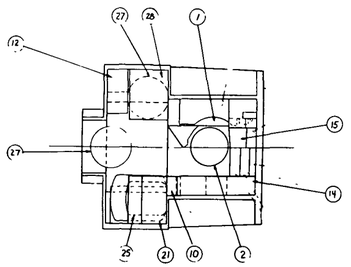
VIEW LOOKING OUTBOARD
LEFT HAND SIDE



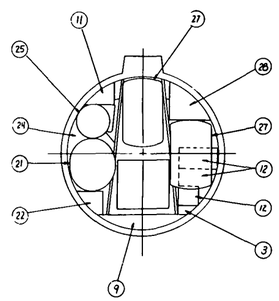
VIEW LOOKING AFT



VIEW LOOKING OUTBOARD
RIGHT HAND SIDE



PLAN VIEW



VIEW LOOKING FORWARD

Figure 2-4. Spacecraft Experiment Configuration, Mission 2, Launch B
(Sheet 2 of 2) (32 pages 1-8.32)

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2-44-2

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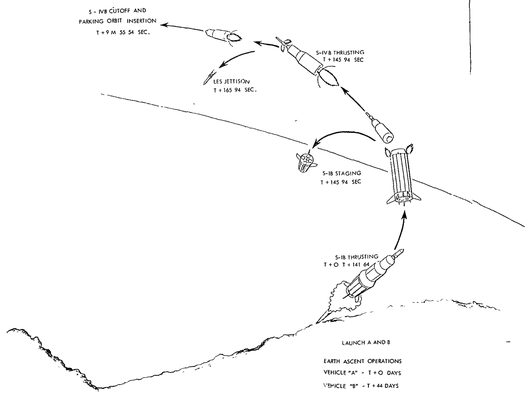
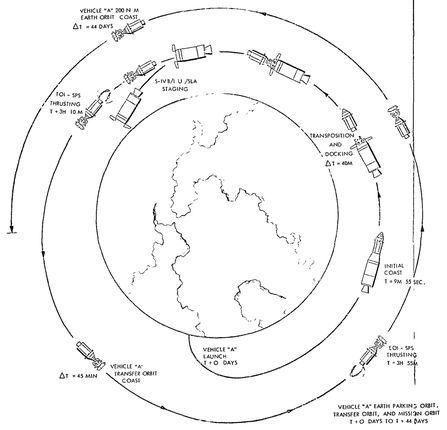
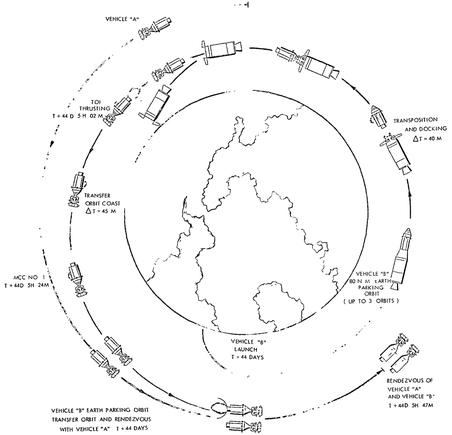
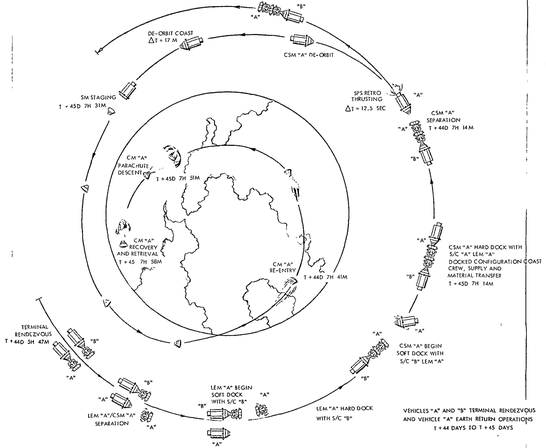
2-44-3





MISSION 2 SA-1B TO SA-1B RENDEZVOUS





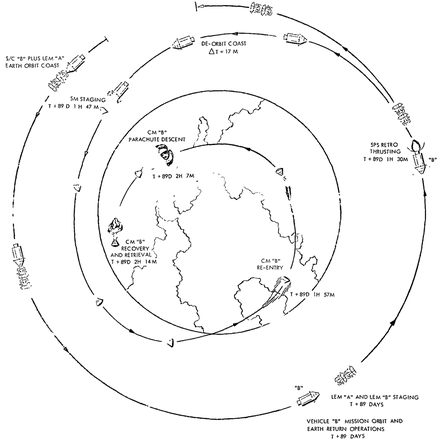


Figure 2-5. Configuration History

2-45-1

2-45-2

2-45-3

2-45-4

2.2.2.5 Expendable Weights, Vehicle A

2.2.2.5.1 General

Expendable weights are summarized for Vehicle A of Mission 2 as shown in Figure 2-6.

2.2.2.5.2 Life Support System

The life support system of Vehicle A will carry the following expendables:

a. Food - 228 pounds are stored on the LEM (see 32-1-7.27). This amount of food is sufficient to sustain the crew for the mission at a consumption rate of 1.4 pounds per man per day (see 126).

b. Oxygen - The two Block II CSM oxygen tanks have a total capacity of 652.8 pounds (see 80-167). The two LEM Apollo X-tanks (see 32-1-7.12) are loaded to 2063 pounds.

c. Hydrogen - The two Block II CSM hydrogen tanks have a total capacity of 58.24 pounds (see 80-167). The LEM has two Apollo X-tanks (see 32-1-7.12) loaded to 169 pounds.

d. Repressurization - For extravehicular activity, the CM atmosphere must be evacuated overboard. It is assumed that the CM and LEM have no airlocks and the LEM atmosphere is also dumped. Repressurization requires six pounds of oxygen for the LEM (see 127), and seven pounds for the CM (see NAA Training Manual). Five repressurizations are required (see 31-1-7.12).

e. Leakage - The leakage rate for the CM is 5.66 pounds per day (see 126); for the LEM, 4.8 pounds per day. This results in a total oxygen leakage of 216 pounds for the LEM and 254 pounds for the CM, based on a cabin pressure of 5 psia.

f. Respiration - The crew oxygen requirement is 1.67 pounds per man per day (see 126), or 225.5 pounds supplied by the LEM.

g. Fuel Cells - The CSM and the LEM each have three fuel cell assemblies (see 80 and 32-1-7.12). The hydrogen and oxygen supply for the CSM cells is the original supply less the amounts required for respiration, repressurization, and leakage replacement.

h. Water - The fuel production of water varies, with power requirements, within 28.8 and 82.8 pounds per day for three fuel cell assemblies. The weight of hydrogen and oxygen available to the fuel cells combines to form water.

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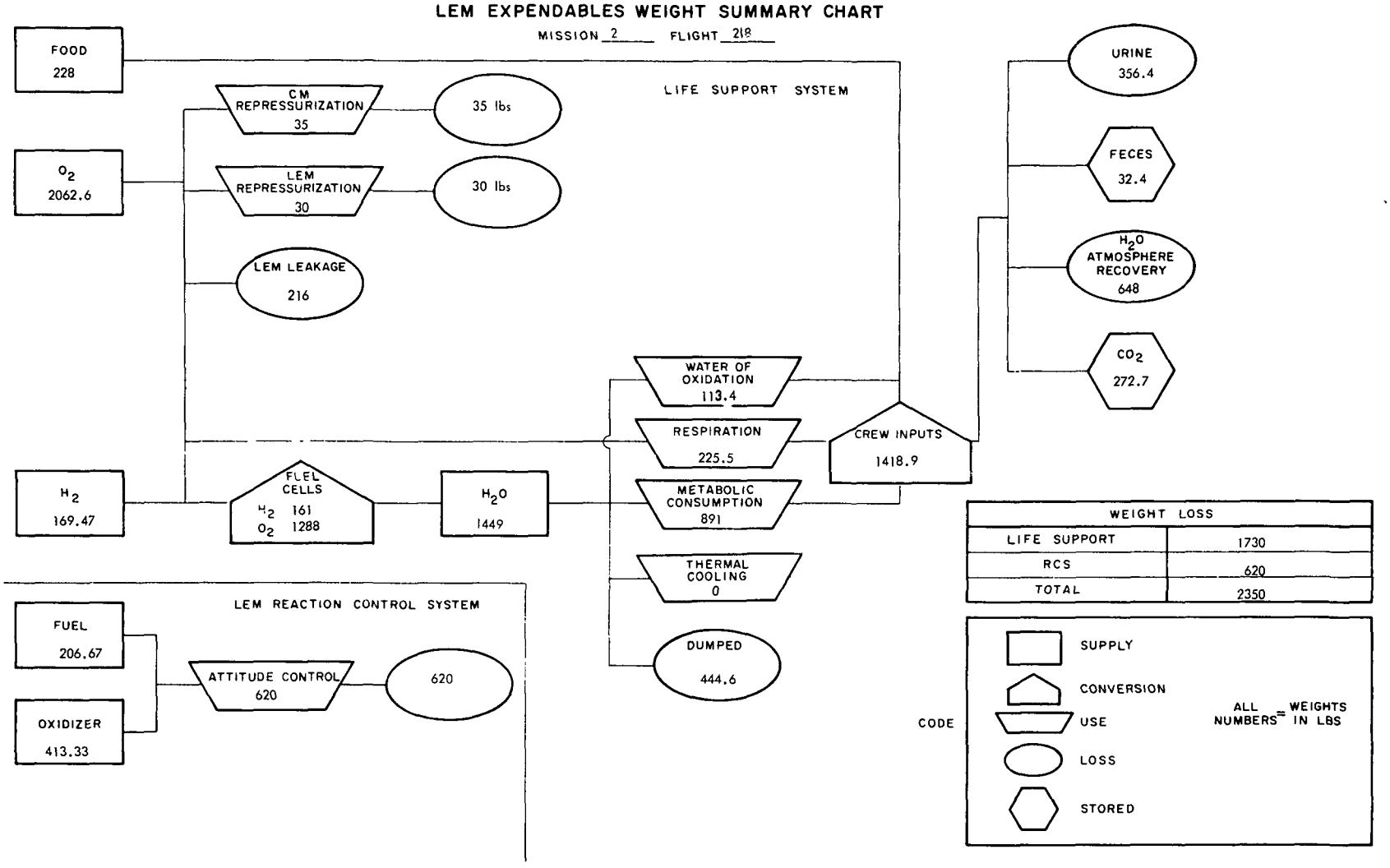

Figure 2-6. Expendable Weights (218 Flight)
(Sheet 1 of 2)

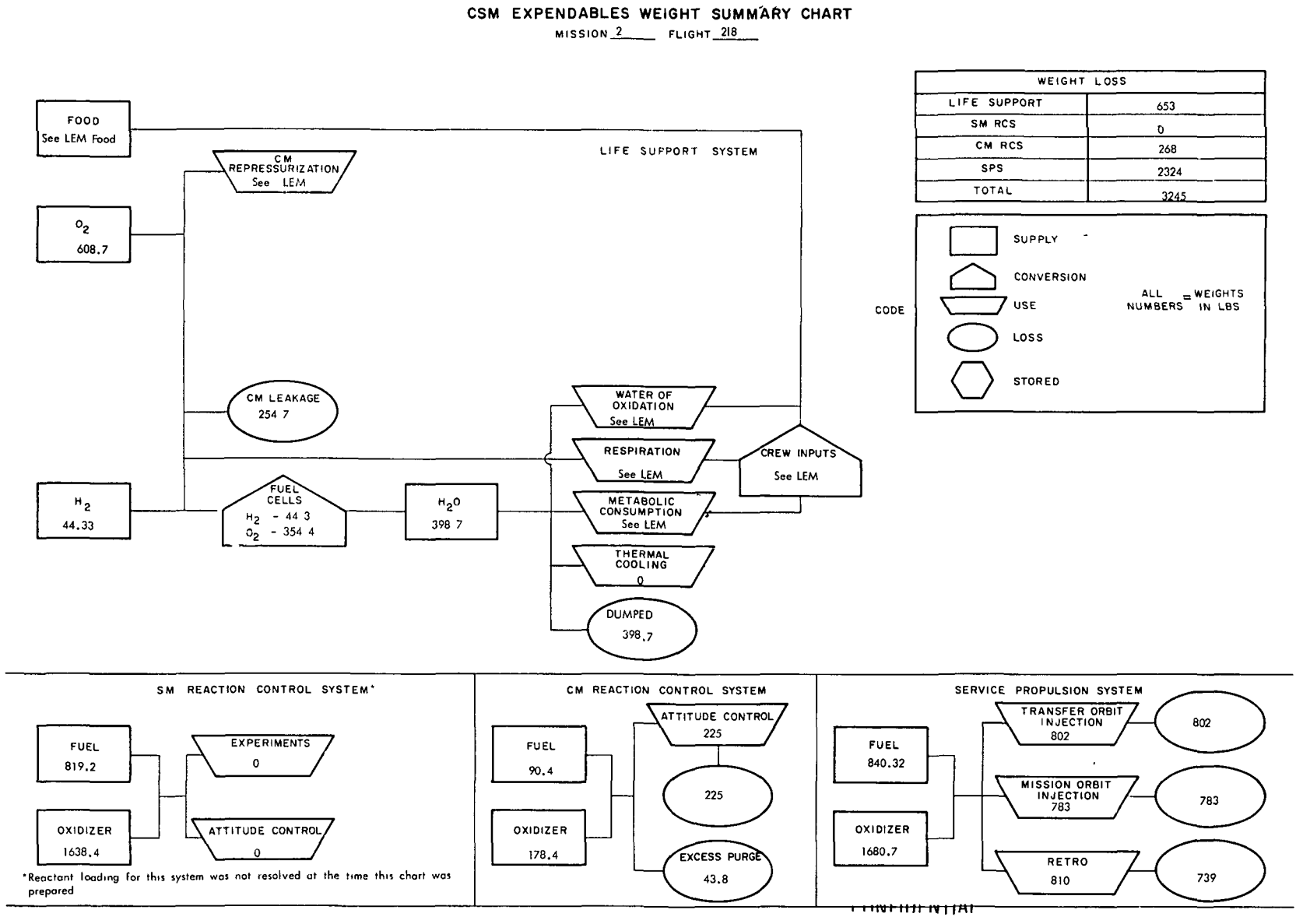

Figure 2-6. Expendable Weights (218 Flight)
(Sheet 2 of 2)

i. Water of Oxidation - Water is needed to reconstitute powdered food. The requirement is 0.035 pounds per man per hour (see 126) for a total of 113 pounds.

j. Metabolic Consumption - Drinking water requirement is 0.275 pounds per man per hour (see 126) for a total of 891 pounds.

k. Thermal Cooling - The LEM cooling water requirement is zero pounds (see 31-1-7.12). It is assumed that the CM requirement will be the same.

l. Water Dumped - Any excess water produced by the fuel cells is vented overboard.

m. Crew Inputs - The following are crew inputs to total weight:

(1) Urine - Produced at a rate of 0.11 pounds per man per hour (see 126), totaling 356 pounds that are vented overboard.

(2) Feces - Produced at a rate of 0.24 pounds per man per day (see 126) totaling 32.4 pounds that are stored aboard.

(3) Water Atmosphere Recovery - Water vapor from expiration and perspiration, given off at a rate of 0.2 pounds per man per hour (see 126). The 648-pound total is recovered and dumped.

(4) Carbon Dioxide - Human respiration waste, produced at a rate of 2.02 pounds per man per day (see SID 63-1370-2). The total amount, 273 pounds, is stored onboard in lithium hydroxide filters.

2.2.2.5.3 SM RCS

The SM RCS fuel and oxidizer tanks (two each per quad) are not filled for this flight (zero pounds) (see Vol. I, Section 2).

2.2.2.5.4 CM RCS

The two Block II CM RCS fuel tanks (see 80-85) are filled to 90.4 pounds. The two Block II CM RCS oxidizer tanks are filled to 178.4 pounds. The 225 pounds of reactants provide:

a. Maneuver capabilities during entry after SM separation.

b. Maneuver capabilities in the event of a high-altitude abort after LES jettison.

c. Three-axis control to stabilize CM motion prior to deployment of the LES parachutes.

Fuel and oxidizer left in the tanks after parachute deployment and final attitude correction is dumped before touchdown.

2.2.2.5.5 SPS

The single Block II SPS fuel tank (see 80-viii) has a total capacity of 15,435 pounds; the load for this flight is 840 pounds. The single Block II oxidizer tank has a capacity of 31,620 pounds; the load for this flight is 1681 pounds.

The reactant supply used to transfer from the 80-n. mi. orbit to the 200-n. mi. orbit weighs 802 pounds (see 111). The reactant used to inject into the 200-n. mi. orbit is 783 pounds (see 111). A reactant weight of 739 pounds is required for the retro maneuver (calculated).

2.2.2.5.6 LEM RCS

The four Block II LEM RCS Fuel tanks (see 31-1-7.12) have a 405-pound total capacity; the load for this flight is 206 pounds. The four Block II oxidizer tanks have an 819-pound total capacity (see 31-1-7.12); the load for this flight is 413 pounds.

The requirement for LEM attitude control is 620 pounds of reactant (Bellcomm, Inc., weight change).

2.2.2.6 Expendables Weights, Vehicle B

2.2.2.6.1 General

The expendable weights for Vehicle B of Mission 2 are summarized in Figure 2-7.

2.2.2.6.2 Life Support System

The following weights are ascribed to expendables of the life support system (see 126 and 127):

a. Food - 228 pounds are stored on the LEM (see 32-1-8.28); this amount is sufficient to sustain the crew for the mission at a consumption rate of 1.4 pounds per man per day.
b. Oxygen - The CSM has two Block II oxygen tanks loaded to 652.8 pounds (see 80-167). The LEM has two Block II oxygen tanks and two Apollo X-tanks loaded to a combined total of 2335 pounds.

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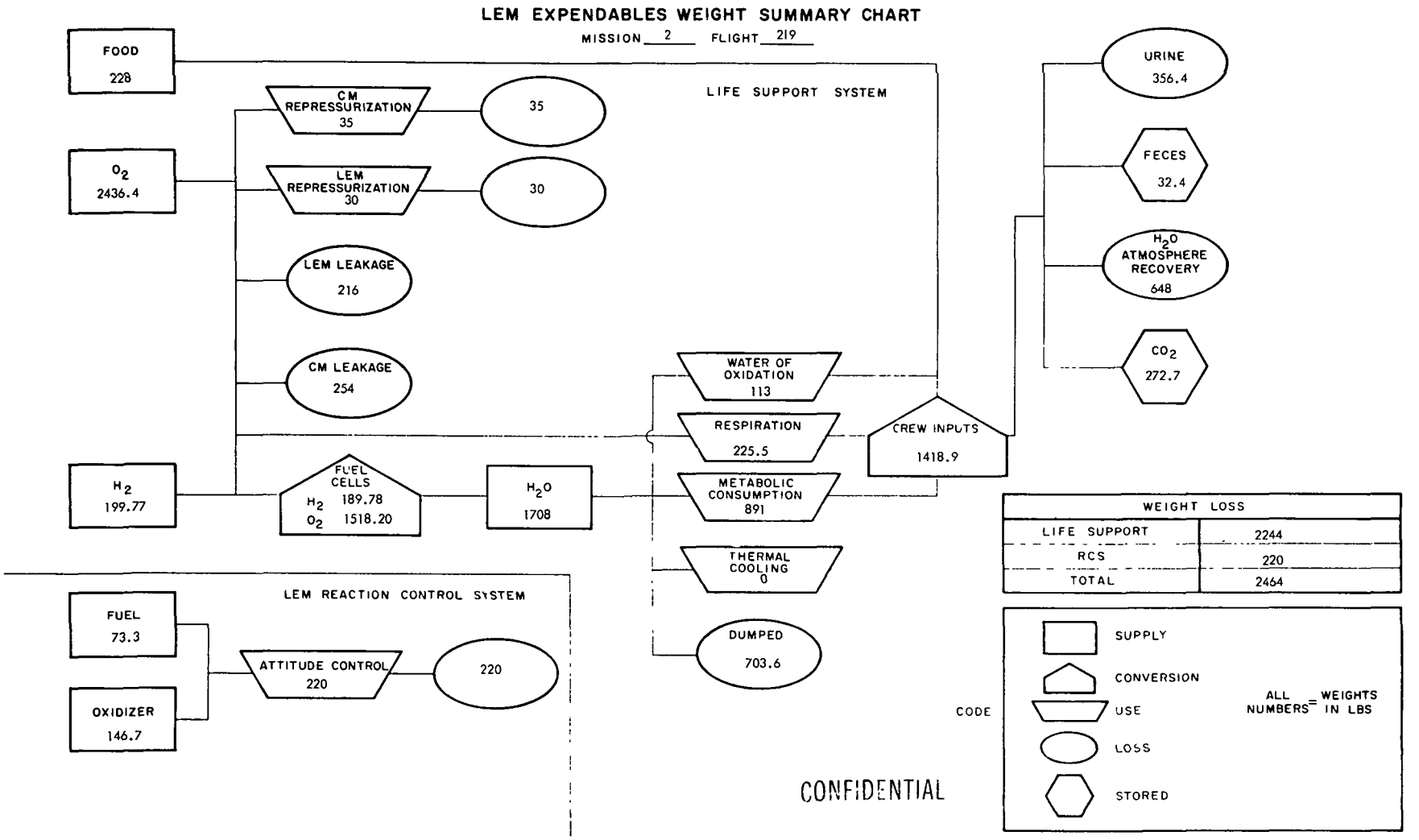

Figure 2-7. Expendable Weights (219 Flight)
(Sheet 1 of 2)

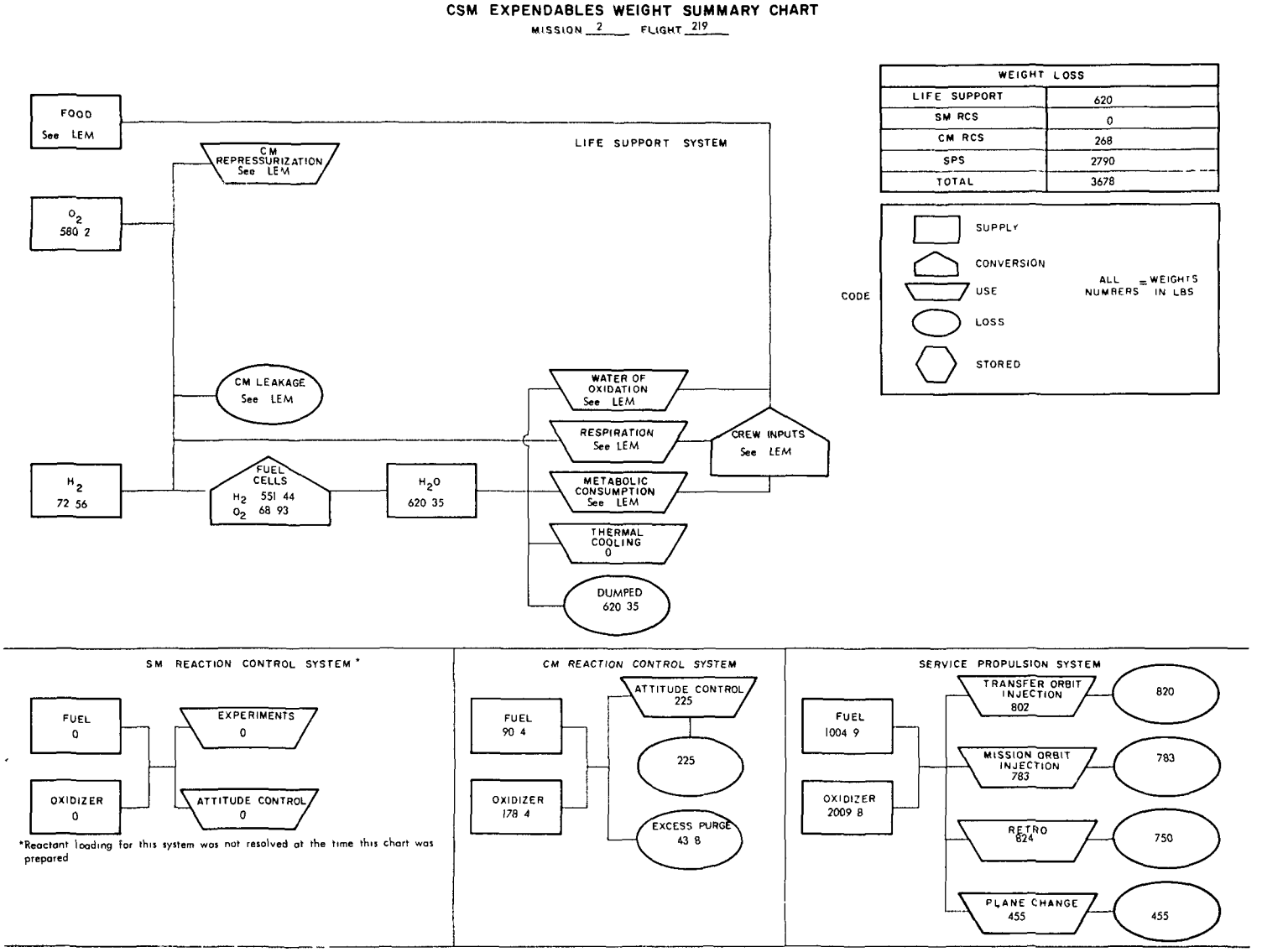

Figure 2-7. Expendable Weights (219 Flight)
(Sheet 2 of 2)

c. Hydrogen - The CSM has two Block II hydrogen tanks loaded to 58.24 pounds (see 80-167). The LEM has two Apollo X-tanks loaded to 210 pounds.

d. Repressurization - For extravehicular activity, the CM atmosphere must be evacuated overboard. It was assumed that the CM and LEM have no airlocks and that the LEM atmosphere is also dumped. Repressurization requires six pounds of oxygen for the LEM, and seven pounds for the CM (NAA Training Manual). Five repressurizations are required (see 32-1-8.13).

e. Leakage - The leakage rate for the CM is 5.66 pounds per day, and for the LEM, 4.8 pounds per day. This results in total oxygen leakage of 254 pounds for the LEM and 216 pounds for the CM. This data is based on a cabin pressure of five psia.

f. Respiration - The crew oxygen requirement is 1.67 pounds per man per day, or 2255 pounds total, to be supplied by the LEM.

g. Fuel Cells - The CSM and the LEM each have three fuel cell assemblies (see 80 and 32-1-8.13). The hydrogen and oxygen supply for the CSM cells is the original supply less the amounts required for respiration, repressurization, and leakage replacement.

h. Water - The fuel cell production of water varies, with power requirements, within 28.8 and 82.8 pounds per day for the three cell-assemblies combined. The total weight of hydrogen and oxygen available combines in the fuel cells to form water.

i. Water of Oxidation - Water is needed to reconstitute the powdered food. The requirement is 0.035 pounds per man per hour for a total of 113.4 pounds.

j. Metabolic Consumption - Drinking water requirement is 0.275 pounds per man per hour for a total of 891 pounds.

k. Thermal Cooling - Cooling water is not required for the LEM (see 32-1-8.13). The same is assumed for the CM.

l. Water Dumped - Excess water produced by the fuel cells is vented overboard.

m. Crew Inputs - The following are crew inputs to total weight:

(1) Urine - Produced at a rate of 0.11 pounds per man per hour, for a total of 356 pounds that are vented overboard.

(2) Feces - Produced at a rate of 0.24 pounds per man per day. The total amount of 32.4 pounds is stored aboard.

(3) Water Vapor - Water vapor of expiration and perspiration, given off at the rate of 0.2 pounds per man per hour. The water, a total of 648 pounds, is recovered and dumped.

(4) Carbon Dioxide - Human respiration waste produced at a rate of 2.02 pounds per man per day (see SID 63-1370-2). The total amount, 272 pounds, is stored onboard in lithium hydroxide filters.

2.2.2.6.3 SM RCS

The SM RCS fuel and oxidizer tanks (two of each per quad) are not filled for this flight.

The SM RCS requirement for attitude control during the experiments is zero pounds of fuel-oxidizer mixture. The reactants are available for normal attitude control function, and for:

a. CSM booster separation.
b. Ullage maneuver prior to SPS ignition.
c. Separation of SM and CM.

2.2.2.6.4 CM RCS

The two Block II CM RCS fuel tanks (see 80-85) are filled to 90.4 pounds. The two Block II CM RCS oxidizer tanks are filled to 178.4 pounds.

Approximately 225 pounds of reactants are required for the following:

a. Maneuvering capabilities during entry after SM separation.
b. Maneuvering capabilities in the event of a high-altitude abort after LES jettison.
c. Three-axis control to stabilize CM motion prior to deployment of ELS parachutes.

Fuel and oxidizer left in the tanks after parachute deployment and final attitude correction is dumped before touchdown.

2.2.2.6.5 SPS

The one Block II SPS fuel tank (see 80-viii) has a total capacity of 15,435 pounds; the fuel load for this flight at liftoff is 1005 pounds. The one Block II SPS oxidizer tank has a 31,620 pound capacity; the load for this flight is 2010 pounds.

For transfer orbit injection, 802 pounds of reactants is required (see 111); for mission orbit injection, 783 pounds of reactant is required. The reactant weight for the retro maneuver (calculated) is 750 pounds, and for plane changes in preparation for rendezvous is 455 pounds.

2.2.2.6.6 LEM RCS

The four Block II LEM RCS fuel tanks (see 32-1-8.13) have 406 pounds total capacity; the load for this flight is 73 pounds. The four Block II LEM RCS oxidizer tanks (see 32-1-8.13) have 819 pounds total capacity; the load for this flight is 147 pounds. LEM attitude control requires 220 pounds of reactant.

2.2.2.7 Weight Profile

2.2.2.7.1 General

The vehicle weight changes for the portion of Mission 2 beginning with S-IVB burnout through CM re-entry are shown in Figure 2-8. Data for this chart was taken from Reference III, from the Weight Summary Tables (Tables 2-1 and 2-2), and from the Expendables Weight Summary Charts (Figures 2-6 and 2-7).

2.2.2.7.2 Flight A

Vehicle A weight at S-IVB burnout is 71,255 pounds. S-IVB staging and jettison of the LEM adapter reduces the weight to 36,270 pounds. Fuel expended for transfer orbit injection and mission orbit injection brings the weight down to 34,685 pounds.

The payload weight required for this flight is 32,655 pounds. The differential between maximum vehicle weight and actual payload weight is shown as "margin," which for this flight is 2030 pounds.

During the 44-day earth orbit coast, the consumption of expendables reduces the spacecraft weight to 29,652 pounds.

WEIGHT PROFILE

MISSION 2 SATURN IB TO SATURN IB RENDEZVOUS

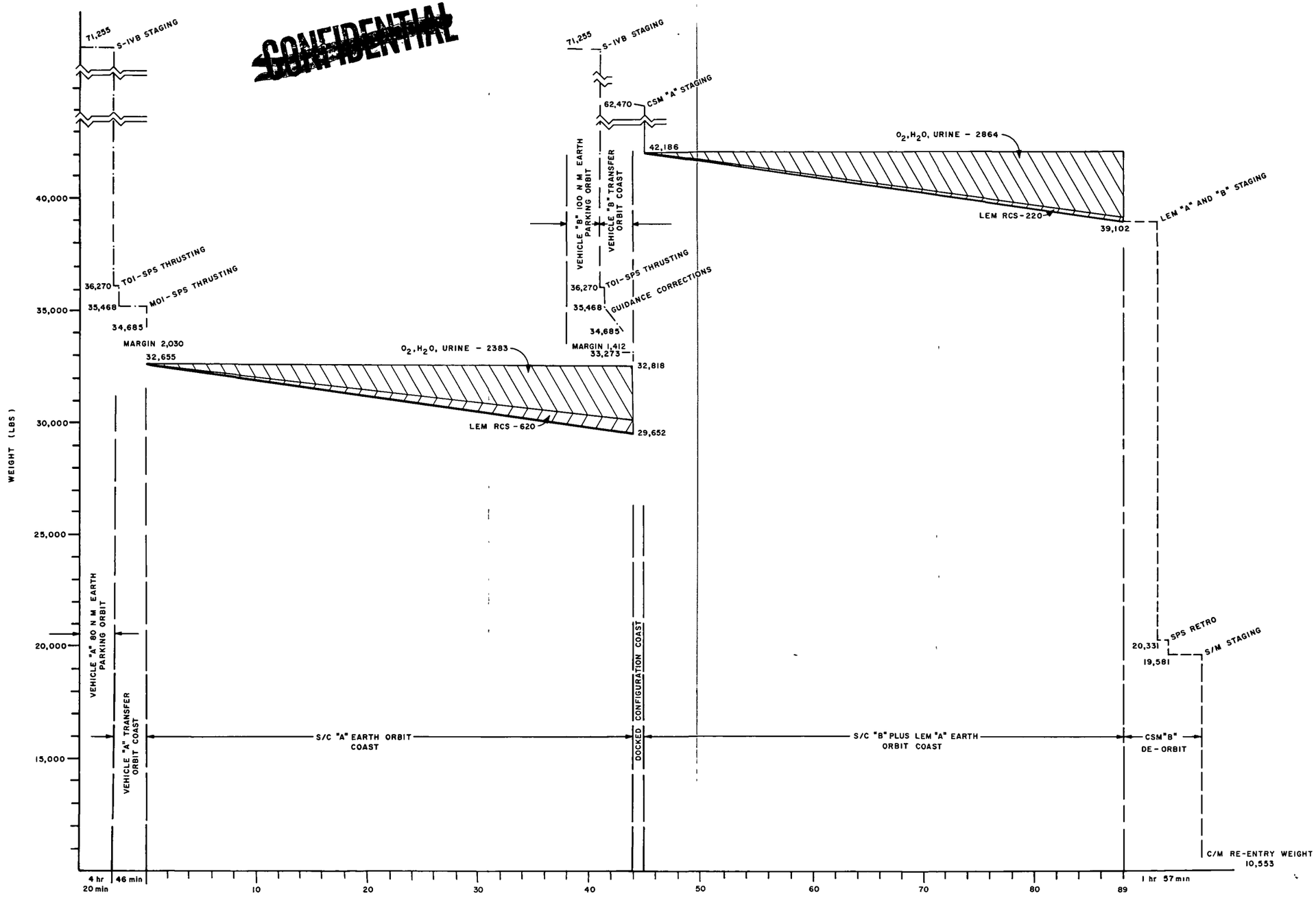

Figure 2-8. Weights Profile

CONFIDENTIAL

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2.2.2.7.3 Flight B

For this flight the vehicle weight at S-IVB burnout is 71,255 pounds. S-IVB staging and jettisoning the LEM adapter reduces the weight to 36,270 pounds.

Fuel expended for transfer orbit injection and midcourse corrections brings the weight down to 34,685 pounds. The payload weight required and the available payload weights are shown as "margin," and are 1413 pounds.

The 33,273 pounds payload is reduced to 32,818 pounds, assuming a maximum allowable plane change maneuver to accomplish rendezvous.

2.2.2.7.4 Vehicles A and B Docked and Re-Entry

The combined docked weight of Vehicles A and B is 62,470 pounds. The CSM of Vehicle A is staged and returns to earth. The re-entry weight of CM A is 10,553 pounds, which assumes transfer of 250 pounds of data to this module. This reduces the payload weight in orbit to 42,186 pounds. During the next 45 days in orbit, the consumption of expendables reduces the weight to 39,102 pounds.

Staging LEM A and LEM B brings the weight to 20,331. Retrofire fuel consumption and SM staging reduce the re-entry weight of CM B to 10,553 pounds.

2.2.3 CREW ANALYSIS

2.2.3.1 Time Utilization

Crew tasks performed during Mission 2 fall into two basic categories: flight operations and experiments. The crew flight operations required from liftoff to the 200-n.mi. orbit and from initiation of de-orbit to touchdown are detailed in Appendix D. Flight operations activities required of the crew during the 200-n. mi. orbit consist primarily of performing system status checks, IMU alignments, and ground communications. Table 2-7 of this report summarizes these system status checks as adapted from the Apollo design reference mission (Reference 25). A detailed task analysis of the crew activities required for conduct of the experiments is also presented in Appendix D.

Analysis of the flight crew operations, and the sleep, eating, hygiene and rest requirements for a typical orbital day indicated that approximately 24 man-hours are available per day to perform experiments. This allows a total of approximately 1080 man-hours available for experiments over the 45 days of Flight 218.* Utilizing the number of experiment sequences and the number of test subjects (see 31, 32, and 33) recommended for the assigned experiments with the experiment times as outlined in Appendix D, it was determined that approximately 1808 man-hours would be required to conduct the assigned experiments. An experiment priority system was established. The priority classifications scheme and priority numbers are as follows (Actual priorities for Flights 218 and 219 are listed in paragraph 2.2.3.3.):

a. First priority experiments are those which satisfy primary mission objectives.
b. Second priority experiments are those which contribute to the assessment of physical health or well being of the astronauts.
c. Third priority experiments are those which satisfy secondary mission objectives.
d. Fourth priority experiments are the biomedical and behavioral experiments which contribute general scientific knowledge.
e. The lowest priority experiments, which have been performed on previous flights, are of such a nature that omission is not detrimental to the mission objectives.

*Since only the first three days of Flight 219 were scheduled, no attempt was made to project that flight for 45 days.

It should be noted that the priority classification scheme and assignments were arbitrarily made due to the excessive experiment time load. They do not constitute a recommended or suggested solution to the experiment priority and integration problem.

The flight operations and experiment scheduling assumptions utilized in scheduling Flights 218 and 219 are outlined in paragraph 2.2.3.2. No communication times for flight operations were programmed on these flights because the communications contact points were not known at the time of scheduling.

The actual schedules for the first 14 days of Flight 218 and 3 days for Flight 219 are presented in Figure 2-9. This 14-day sample presents the pattern of experiment sequences that can be expected for a complete 45 days. Figure 2-10 includes the schedules for the first three days of Flight 219. Tables 2-8 and 2-9 provide for Flights 218 and 219 the total amount of time scheduled for each of the experiments, the number of experiment sequences actually scheduled, and the number recommended (31, 32, and 33) for a 45-day flight.

The time utilization chart shown below shows the percentage of time actually scheduled for the 4 major crew activities and presents the total man-hours of experiments scheduled. Approximately 43 percent of the recommended experiment time was actually scheduled for Flight 218.

Mission 2A Time Utilization

Experiments	24.1 percent
Flight Operations	10.1 percent
Personal Time	54.3 percent
Unscheduled Time	7.2 percent
Experiment Time Recommended*	1808.0 man-hours
Experiment Time Scheduled	777.0 man-hours

*Grumman number of experiment sequences and subjects tested (see 31, 32, and 33), experiment times from Appendix D.

2.2.3.2 Flight Operations Scheduling Assumptions

The astronaut day will consist of 10 hours on duty and 14 off duty; of the 14 hours, six will be devoted to sleep, five to rest, and three to eat and for personal hygiene (see 31, 32, and 33).

Three IMU alignments are required each 24 hours.

Systems status checks for flight operations require 4.75 man-hours during a 24-hour period (AMPTF adapted) and are detailed in Table 2-7. Any system checks required for experiments are included in the experiment setup times.

2.2.3.3 Experiment Scheduling Assumptions

Artificial light is assumed to be available for experiments in the 1500-series.

2.2.3.4 Experiment Priorities

2.2.3.4.1 General

The following paragraphs detail the experiment priorities for Flights A and B.

2.2.3.4.2 Flight A Priorities

Priority	Experiment Number and Name	
1	0100 Series	Biomedical - 0102, 0104, 0105, 0106, 0107, 0109, 0110, 0111, 0112, 0113, 0114, 0117, 0118, 0119, 0121.
2	0501	Radiation Environment Monitoring
3	1500 Series	EVA's, Space Technology - 1501, 1502, 1503, 1504, 1505
3	0600 Series	Zero-g, Gases and Liquids - 0601, 0602A, 0602B, 0603A, 0603B, 0604, 0605, 0606.
4	0200 Series	Behavioral - 0201, 0202, 0203.
4	0100 Series	Biomedical - 0108, 0115, 0116, 0120

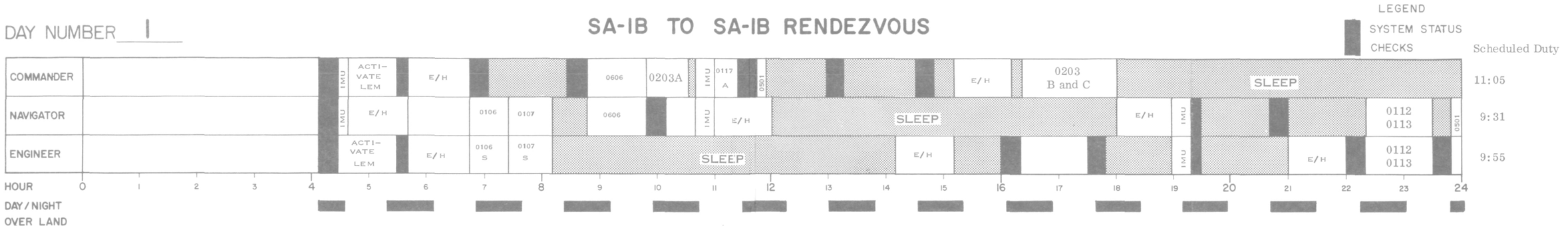
2.2.3.4.3 Flight B Priorities

Priority	Experiment Number and Name	
1	0400 Series	Living Organisms - 0401, 0402, 0403, 0404, 0405, 0406.
2	0501	Radiation Monitoring
3	0302	Onboard Centrifuge

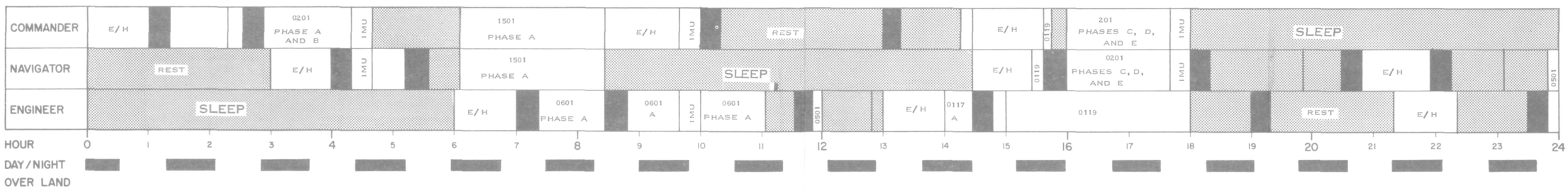
MISSION NUMBER 2A

DAY NUMBER 1

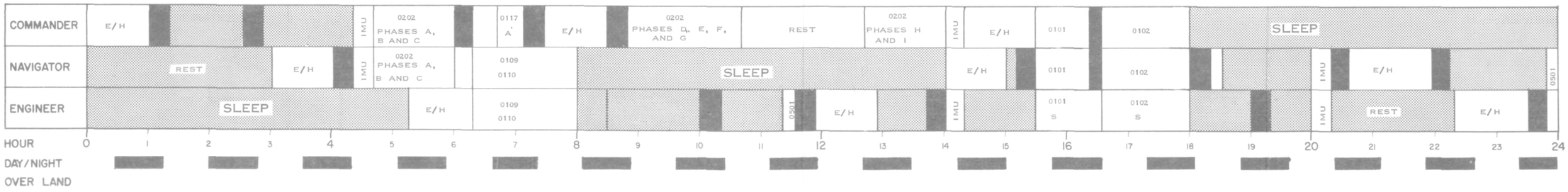
SA-IB TO SA-IB RENDEZVOUS



DAY NUMBER 2



DAY NUMBER 3



NOTE: All Shaded Areas Not Marked Are Rest Periods.

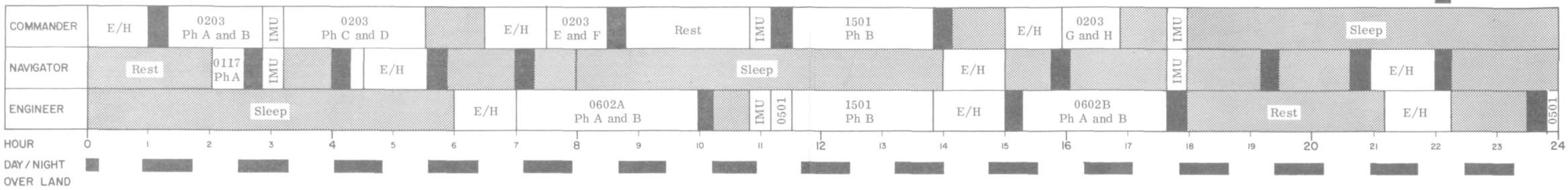
Figure 2-9. Integrated Crew Schedule for First Launch (Sheet 1 of 5)

MISSION NUMBER 2A

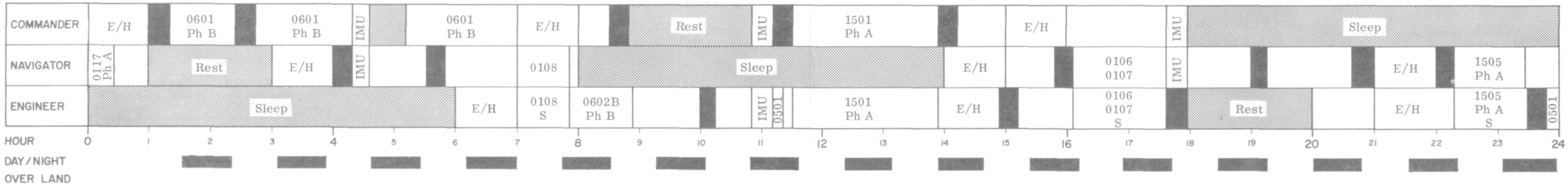
DAY NUMBER 4

SA-IB TO SA-IB RENDEZVOUS

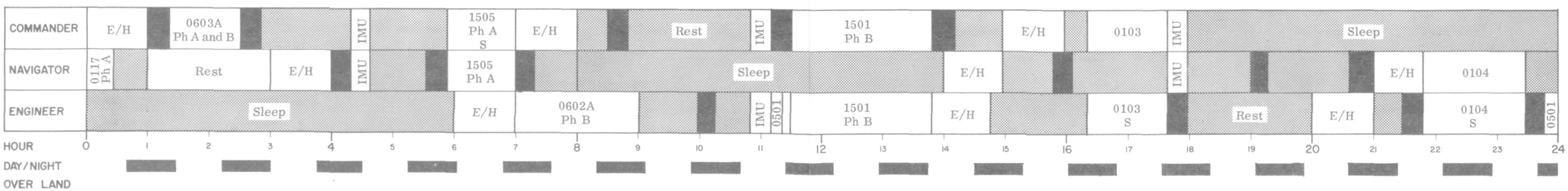
SYSTEM STATUS CHECKS



DAY NUMBER 5



DAY NUMBER 6



NOTE: All Shaded Areas Not Marked Are Rest Periods.

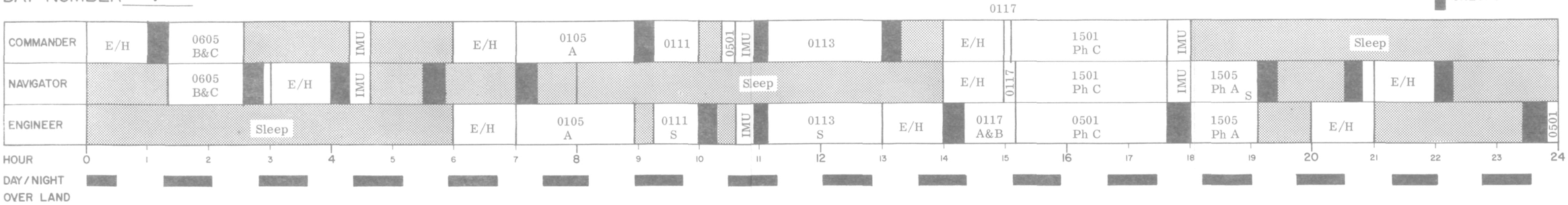
Figure 2-9. Integrated Crew Schedule for First Launch (Sheet 2 of 5)

MISSION NUMBER 2A

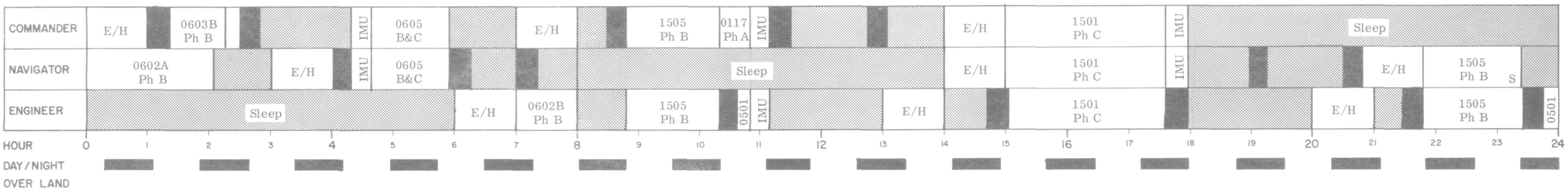
DAY NUMBER 7

SA-IB TO SA-IB RENDEZVOUS

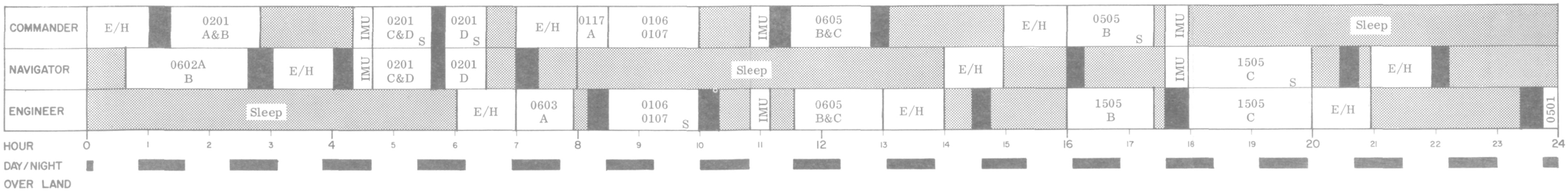
SYSTEM STATUS CHECKS



DAY NUMBER 8



DAY NUMBER 9



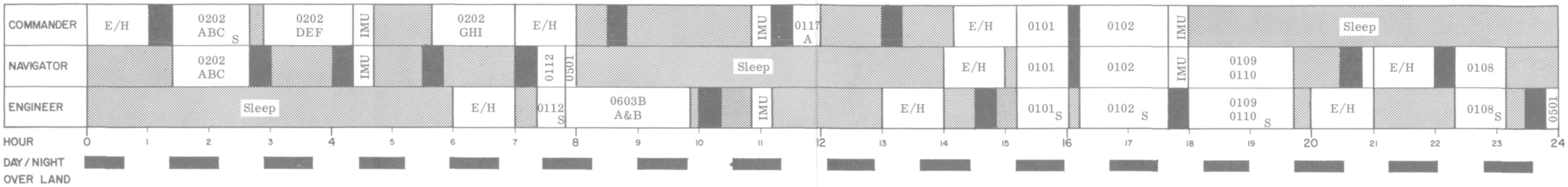
NOTE: All Shaded Areas Not Marked Are Rest Periods.

Figure 2-9. Integrated Crew Schedule for First Launch (Sheet 3 of 5)

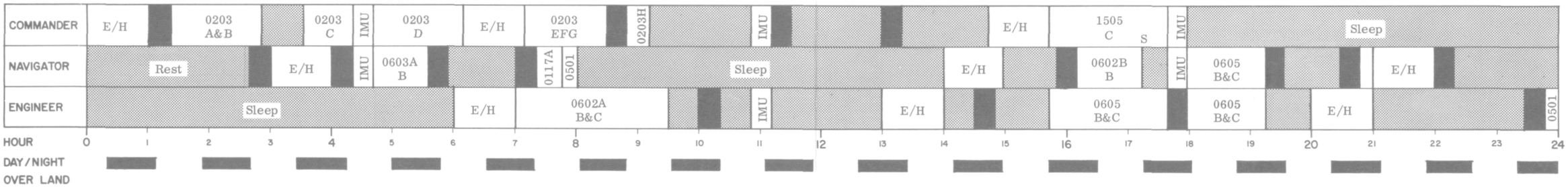
2-65-1

MISSION NUMBER 2A

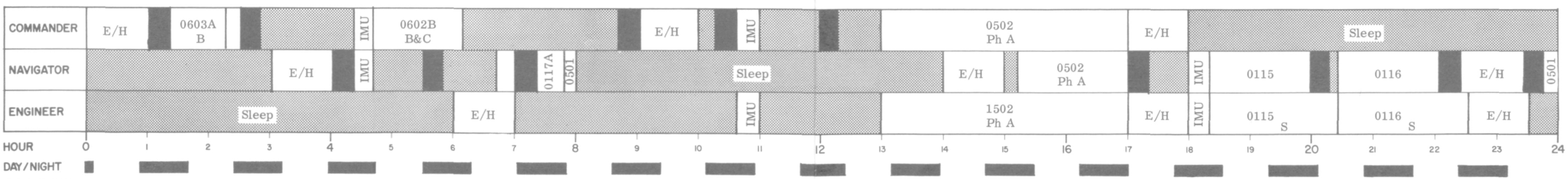
DAY NUMBER 10



DAY NUMBER 11



DAY NUMBER 12



NOTE: All Shaded Areas Not Marked Are Rest Periods.

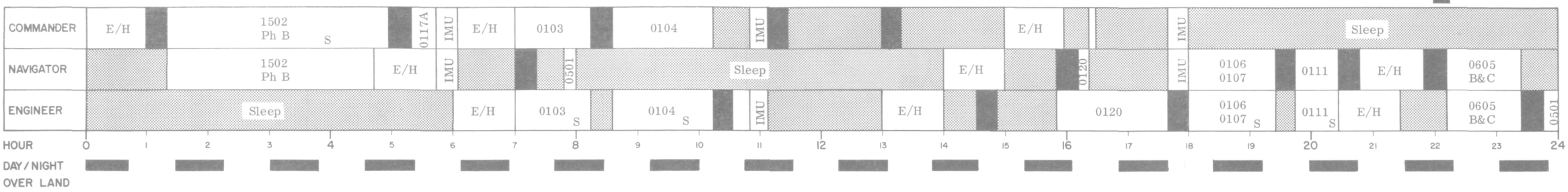
Figure 2-9. Integrated Crew Schedule for First Launch (Sheet 4 of 5)

MISSION NUMBER 2A

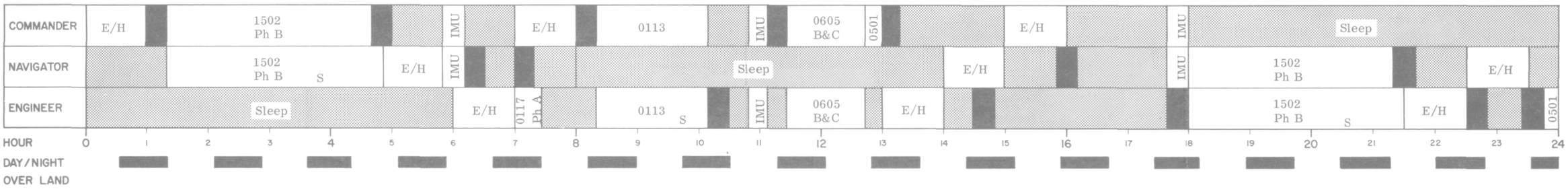
DAY NUMBER 13

SA-IB TO SA-IB RENDEZVOUS

LEGEND: SYSTEM STATUS CHECKS

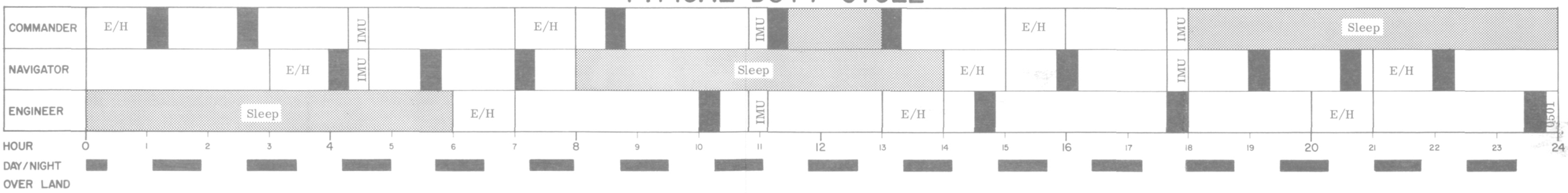


DAY NUMBER 14



DAY NUMBER _____

TYPICAL DUTY CYCLE



NOTE: All Shaded Areas Not Marked Are Rest Periods.

Figure 2-9. Integrated Crew Schedule for First Launch (Sheet 5 of 5)

2-67.1

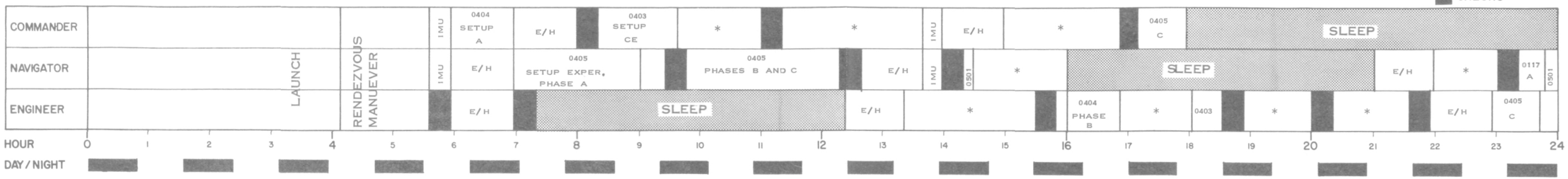
MISSION NUMBER 2B

SA-IB TO SA-IB RENDEZVOUS

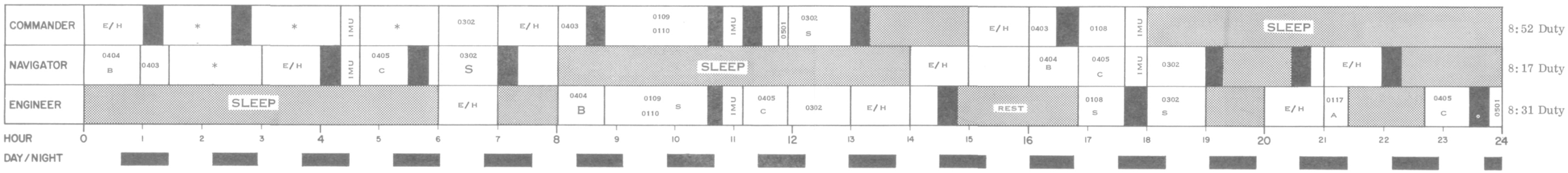
*These periods are not scheduled due to overlap of Missions 2A and 2B.

LEGEND
SYSTEM STATUS CHECKS

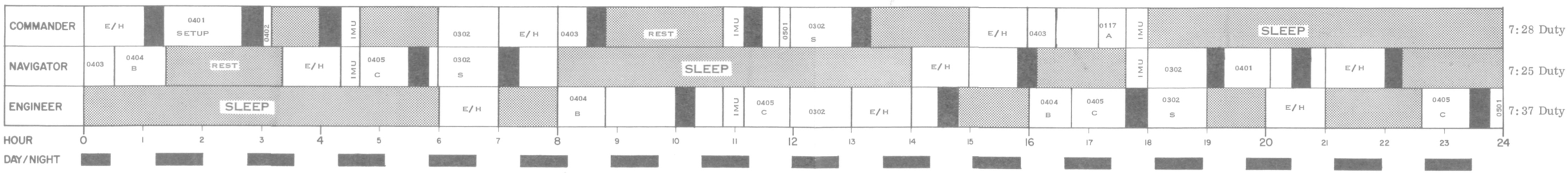
DAY NUMBER 44



DAY NUMBER 45



DAY NUMBER 46



NOTE: All Shaded Areas Not Marked Are Rest Periods.

Figure 2-10. Integrated Crew Schedule for Second Launch

2-68-2

Priority	Experimental Number and Name	
4	0100 Series	Biomedical - 0103, 0104, 0105, 0106, 0107, 0109, 0110, 0111, 0112, 0113, 0114, 0117, 0118, 0119, 0121, 0101, 0102, 0108, 0115, 0116, 0120.

Table 2-7

Breakdown of
Earth Orbital System Status Checks*

Task	Frequency of Performance	Duration (Minutes)
CM		
Check ECS	Hourly	1
Check EPS	Hourly	3
Check SM RCS	Every 4 hours	4
Check CM RCS	Every 12 hours	4
Replace CO_2 and odor absorber	Every 12 hours	8
Charge entry and post-landing batteries	Every 6 hours	3
Purge fuel cells	Every 8 hours	3
Check SPS	Every 4 hours	4
LEM		
Check ECS	Hourly	1
Check PGS	Hourly	1
Check PDS	Hourly	1
Replace CO_2 and odor absorber	Every 12 hours	8
Check LEM RCS	Every 12 hours	4

4.75 man-hours required for system status checks in a 24-hour day.

*AMPTF adapted (31, 32, 33, 24, 25, and 26)

CM tasks were adapted from translunar and transearth coast phase (25).

LEM tasks were adapted from lunar surface phase (25).

Table 2-8

Flight A
Summary of Scheduled Experiment Times and Sequences

Experiment Number	Man-Minutes Expended (14 Days)	Frequency of Performance	
		Actual*	Recommended**
0101	266	2	15
0102	496	2	15
0103	292	2	12
0104	400	2	12
0105	234	1	6
0106	312	4	17
0107	384	4	17
0108	192	2	9
0109	404	2	15
0110			
0111	164	2	9
0112	46	1	9
0113	398	2	55
0114	-	0	36
0115	222	1	18
0116	217	1	4
0117A	350	14	45
B	35	1	6
0118	199	1	5
0119	190	1	5
0120	83	1	9
0121	-	0	2
0201A	108	2	15
B	64	2	7
C	152	2	15
D	84	2	7
E	152	2	7
0202A	84	2	15
B	128	2	15
C	92	2	15

Table 2-8

Flight A
Summary of Scheduled Experiment Times and Sequences (Cont.)

Experiment Number	Man-Minutes Expended (14 Days)	Frequency of Performance	
		Actual*	Recommended**
0202D	96	2	15
E	112	2	15
F	128	2	15
G	48	2	15
H	56	2	15
I	56	2	15
0203A	82	2	15
B	96	2	15
C	98	2	15
D	172	2	7
E	54	2	7
F	56	2	15
G	62	2	15
H	42	2	15
0501	280	28	90
0601A	180	1	1
B	255	1	1
0602AA	60	1	1
B	600	5	5
C	25	1	1
0602BA	40	1	1
B	300	5	5
C	25	1	1
0603AA	15	1	1
B	220	4	10
C	-	0	1
0603BA	30	1	1
B	90	1	3
C	-	0	1
0604A	-	0	1

Table 2-8

Flight A
Summary of Scheduled Experiment Times and Sequences (Cont.)

Experiment Number	Man-Minutes Expended (14 Days)	Frequency of Performance	
		Actual*	Recommended**
0604B	-	0	2
C	-	0	1
0605A	41	1	1
BC	450	6	20
D	-	0	1
0606A	171	1	1
1501A	548	2	2
B	548	2	2
C	842	2	2
1502A	586	1	1
B	1230	3	10
C	-	0	6
1503	-	0	12
1504A	-	0	2
B	-	0	2
C	-	0	2
1505A	330	3	9
B	528	3	9
C	456	2	9
D	-	0	9

* Actual number of sequences scheduled for 14 days (General Electric).
**Recommended number of sequences for 45 days (31, 32, and 33).

Table 2-9

Flight B
Summary of Scheduled Experiment Times and Sequences

Experiment Number	Man-Minutes Expended	Frequency of Performance	
		Actual*	Recommended**
0101	-	0	15
0102	-	0	15
0103	-	0	12
0104	-	0	12
0105	-	0	6
0106	-	0	17
0107	-	0	17
0108	96	1	9
0109 } 0110	202	1	15
0111	-	0	9
0112	-	0	9
0113	-	0	5
0114	-	0	36
0115	-	0	18
0116	-	0	4
0117A	75	3	45
B	-	0	6
0118	-	0	5
0119	-	0	5
0120	-	0	9
0121	-	0	2
0302	732	6	270
0401	91	1	6
0402A	41	1	80
B	55	1	40
C	29	1	40
0403-I	50	1	1
-II	196	7	135
-III	-	0	1

Table 2-9

Flight B
Summary of Scheduled Experiment Times and Sequences (Cont.)

Experiment Number	Man-Minutes Expended	Frequency of Performance	
		Actual*	Recommended**
0404-I	15	1	1
-II,III	364	7	135
-IV	-	0	1
0405-I	120	1	1
-II	102	1	45
-III	495	11	180
-IV	-	0	1
0406-I	-	0	1
-II	-	0	1
-III	-	0	4
-IV	-	0	4
0501	60	6	90

* Total number of sequences scheduled in 3 days.
**Total number of sequences planned for 45 days.

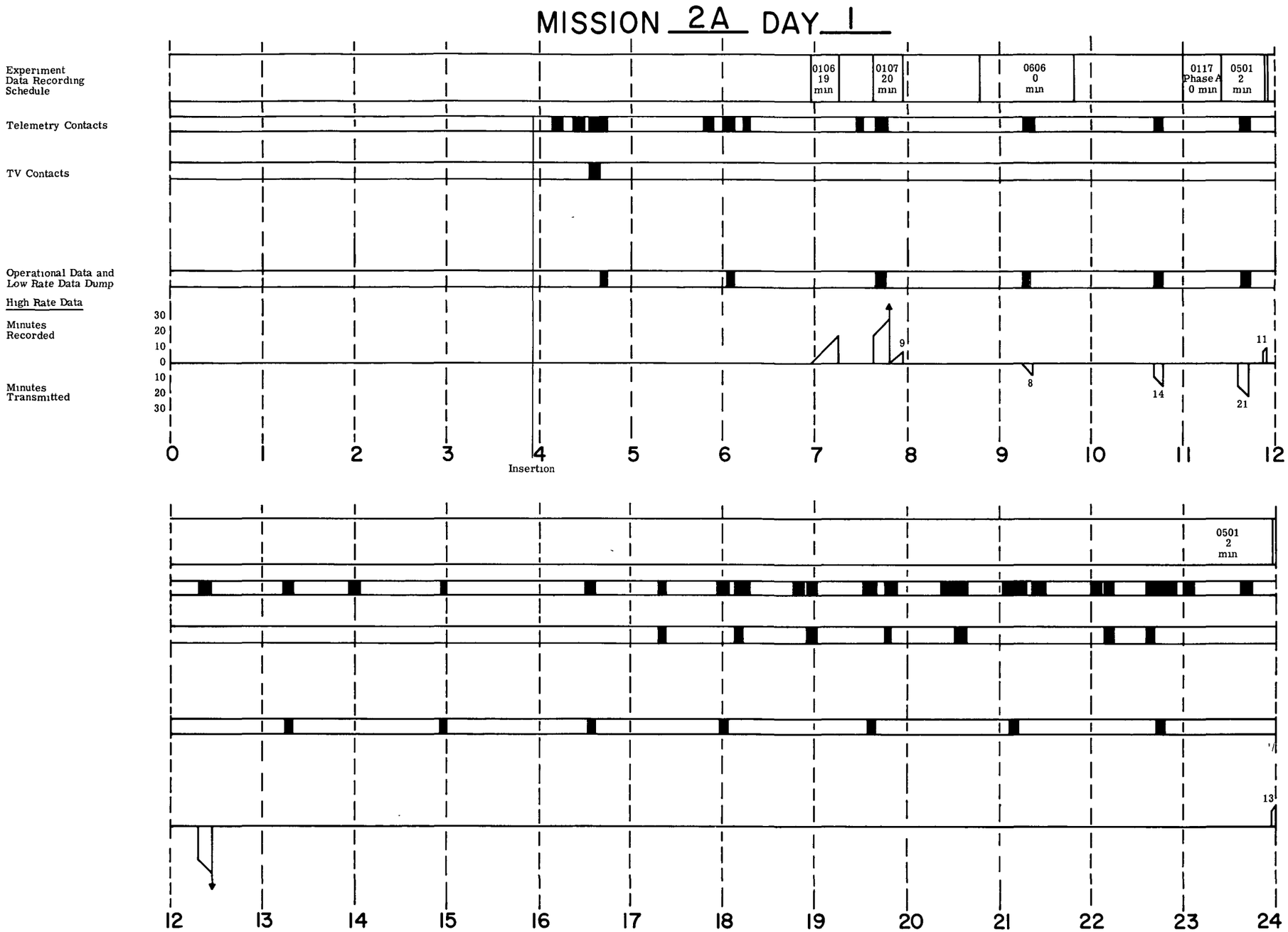
2.2.4 DATA FLOW

2.2.4.1 General

Data flow characteristics - data recording and transmission, magnetic tape storage, and data collection - are detailed in the following paragraphs. The data flow was profiled for Mission 2 using the analysis presented in Part II, Section 3, of this report. Instructions for using the profiles are included in the section under the applicable heading.

2.2.4.2 Data Recording and Transmission

A data recording and transmission profile for the first three days of each flight of Mission 2 is presented in Figures 2-11 and 2-12. A sample profile was developed for each flight of the mission to illustrate graphically the distribution of recording and transmission times.



DATA PROFILE CHART

Figure 2-11. Data Recording and Transmission Profile First Launch (Sheet 1 of 3)

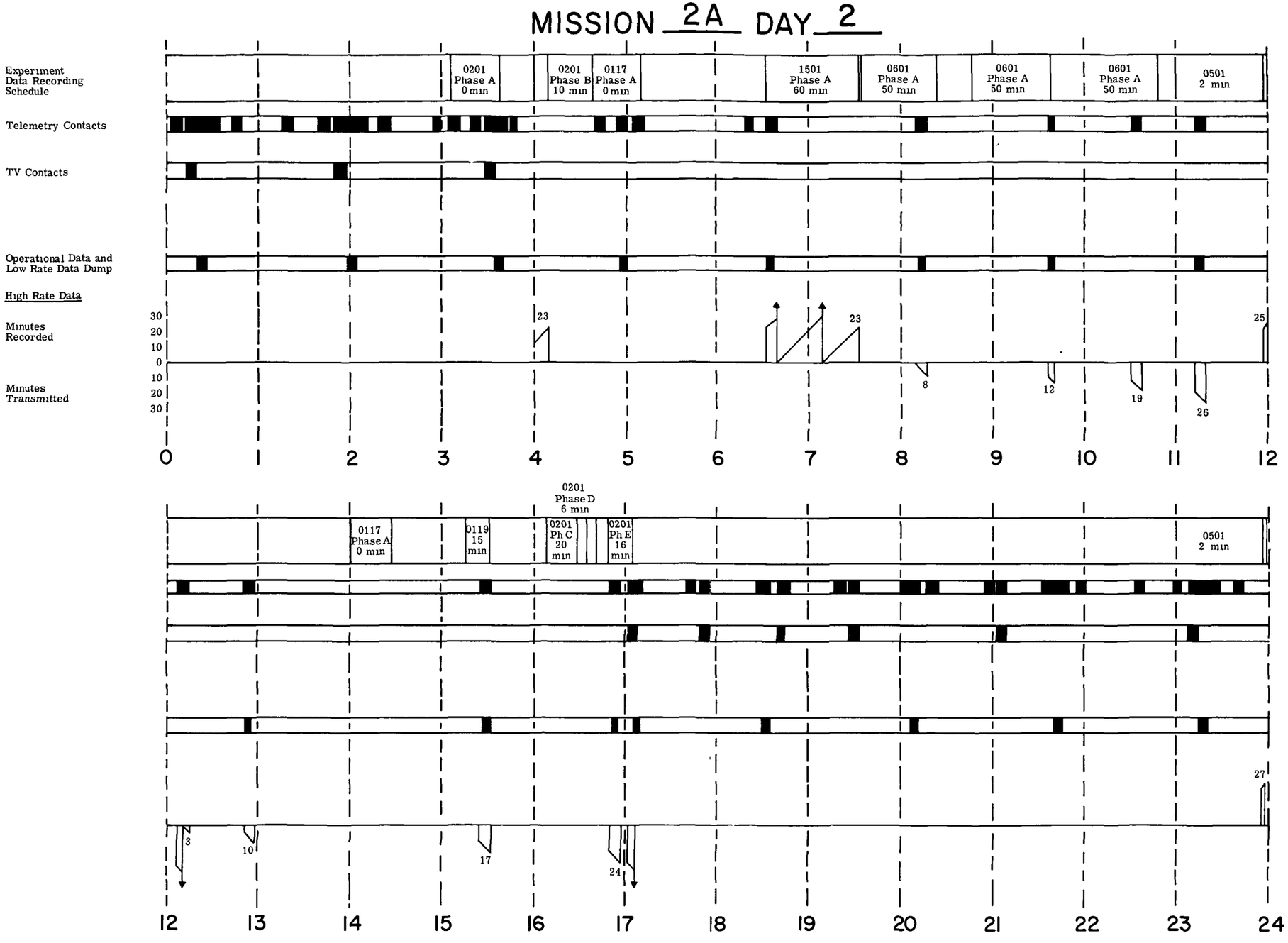


Figure 2-11. Data Recording and Transmission Profile First Launch (Sheet 2 of 3)

2-76-2

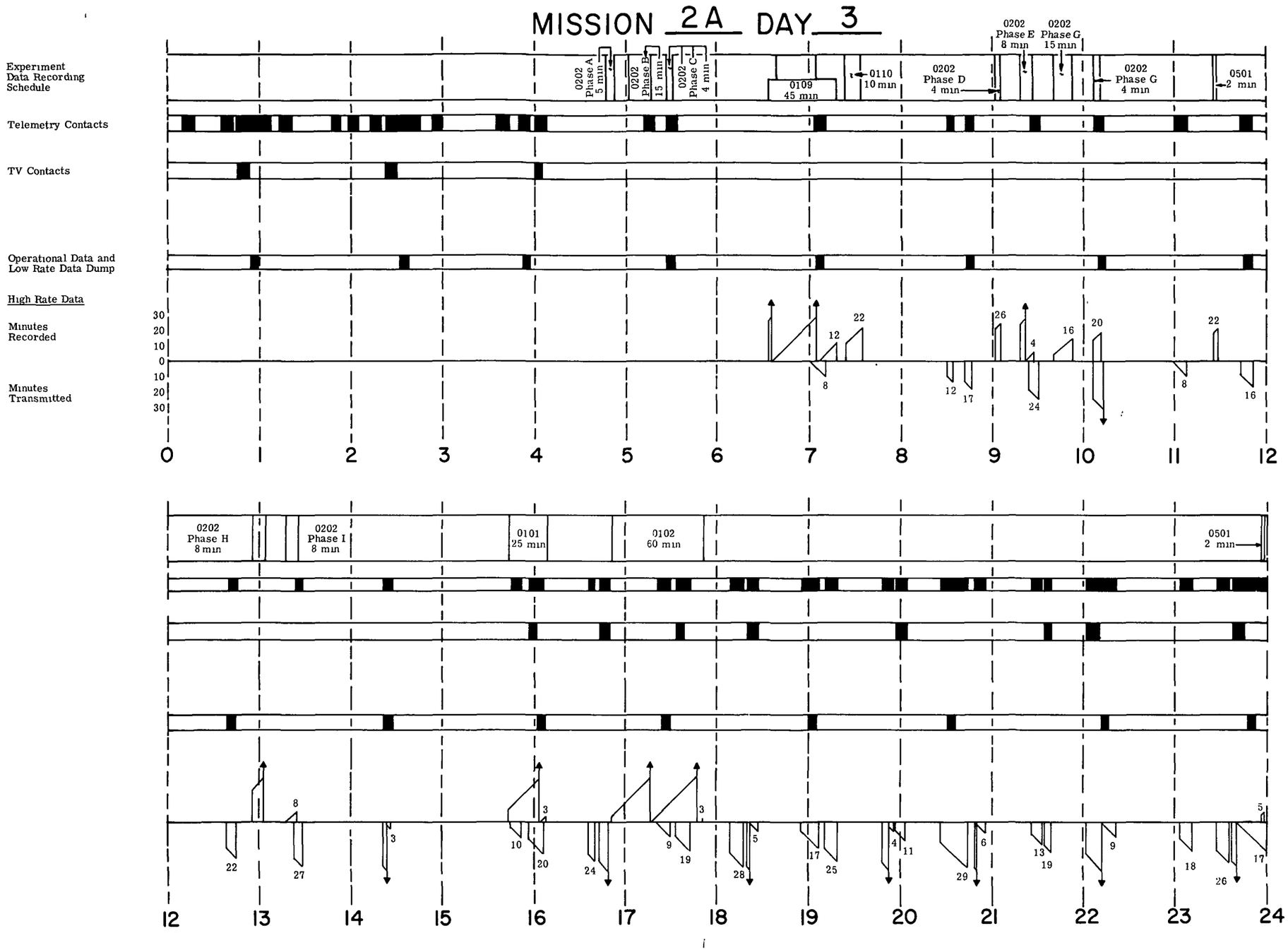


Figure 2-11. Data Recording and Transmission Profile First Launch (Sheet 3 of 3)

2-77-1

2

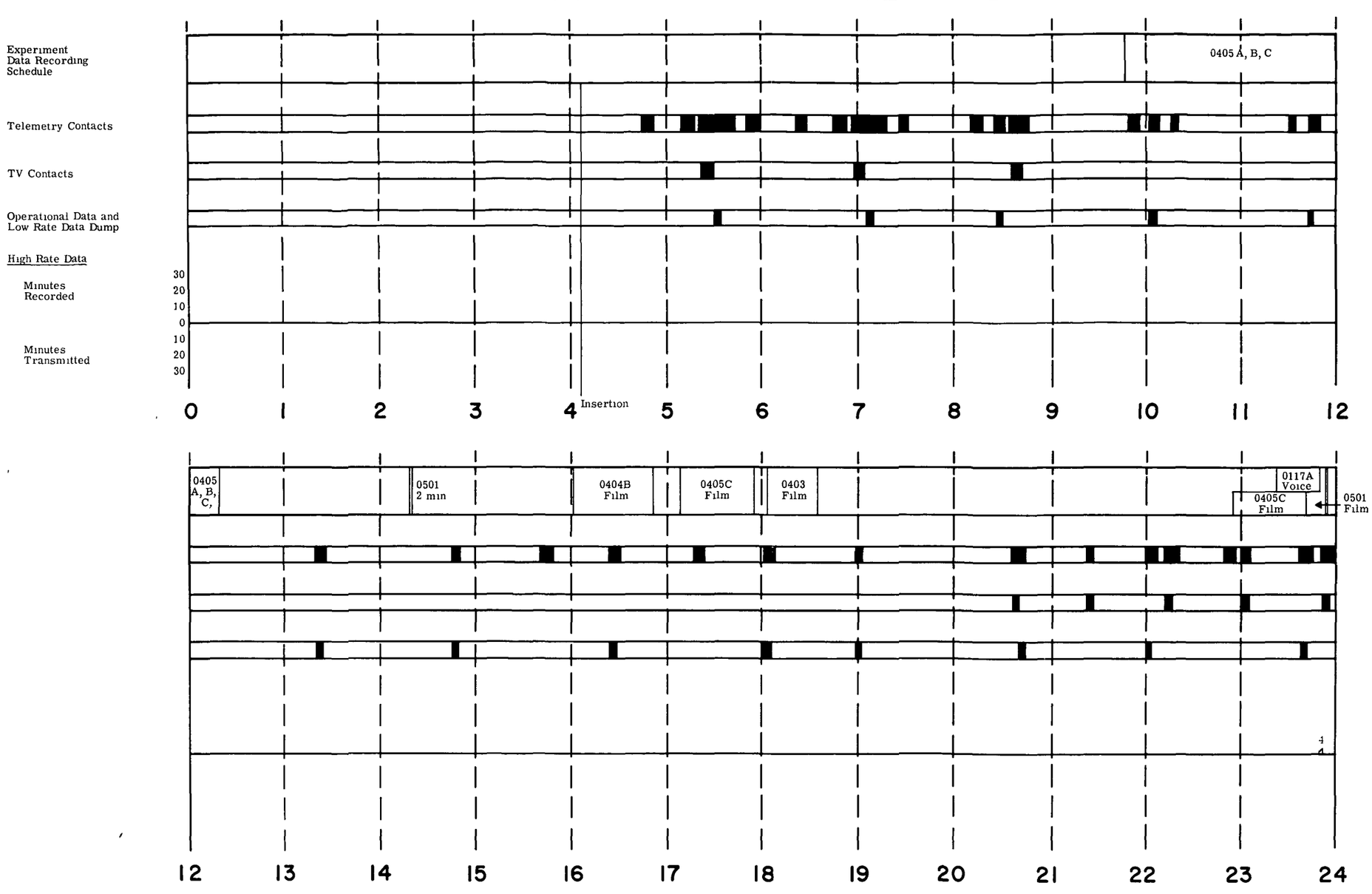
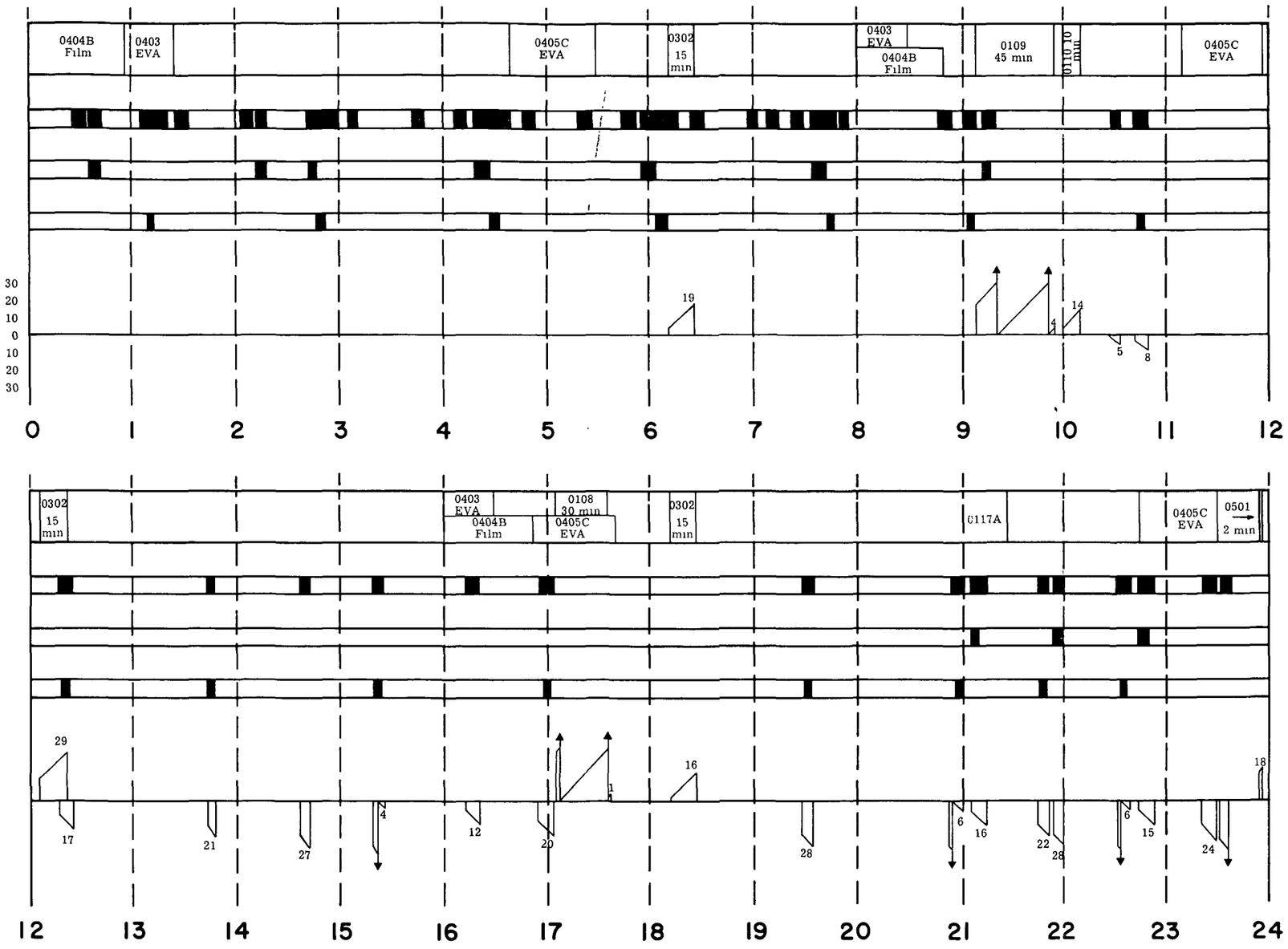


Figure 2-12. Data Recording and Transmission Profile Second Launch (Sheet 1 of 3)

MISSION 2B DAY 2

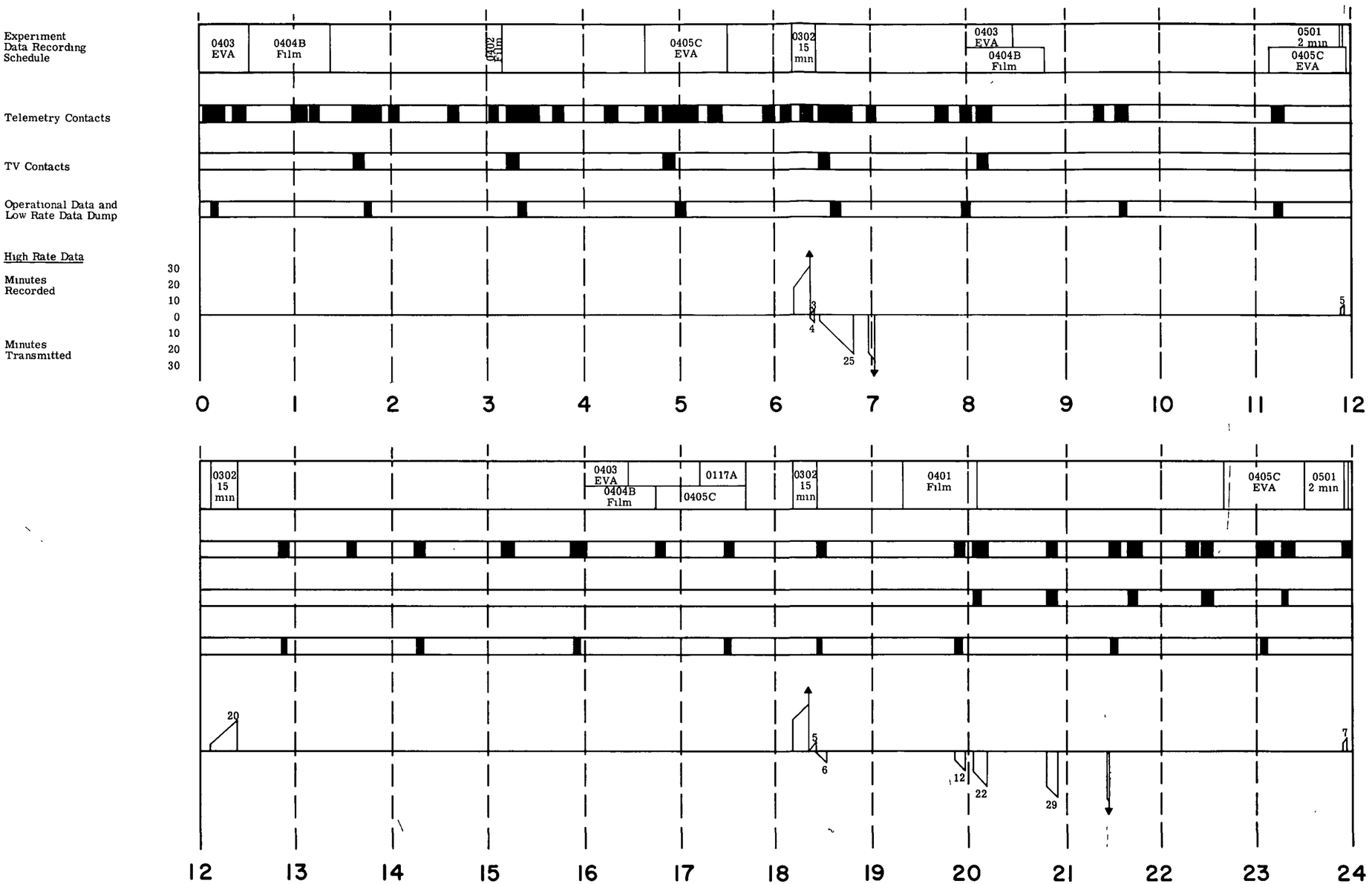


DATA RECORDING AND TRANSMISSION PROFILE

Figure 2-12. Data Recording and Transmission Profile Second Launch (Sheet 2 of 3)

2-79-1

MISSION 2B DAY 3



DATA RECORDING AND TRANSMISSION PROFILE

Figure 2-12. Data Recording and Transmission Profile Second Launch (Sheet 3 of 3)

2-80-2

The top line shows the distribution of experiment data collection sequences and their duration in time. The second line shows the distribution of MSFN S-band telemetry ground station coverages and their durations. The third line shows the coverage distribution of those telemetry stations assumed to have TV receiving capability (Goldstone, Madrid, and Canberra). The fourth line shows the distribution of station contacts tentatively assigned for the reception of dumped operational data (and low-sampling-rate experimental data) from the CSM recorder.

NOTE

These contacts were assigned on a one-dump-per-orbit basis with no interval greater than 120 minutes whenever possible. When intervals greater than 120 minutes occur, an extra operational contact is tentatively assigned to allow the dumping of the extra reel of stored operational data tape. Whenever a choice is available, the operational contact is made over the U. S.

The fifth line is a profile of the rate of accumulation of the magnetic tape being used to record high-rate PCM data in the LEM, compared to the rate at which reels of this tape may be dumped to the ground telemetry stations. Each upward-pointing arrow indicates the accumulation of one full reel of tape (30 minutes recording time) and each downward-pointing arrow indicates the completed dump of one full reel of tape (30 minutes dumping time).

The profile for both flights of Mission 2, shows very little recording activity, well within the presently planned spacecraft data transmission capabilities.

In generating the transmission coverage profiles, it was determined that a non-coverage interval greater than 120 minutes can be expected on the average of once every three days with the presently planned network of 14 S-band telemetry ground stations.

2.2.4.3 Magnetic Tape Storage Profile

For each applicable mission a sample magnetic tape storage profile was developed for the first three days of that mission to show the nature of the net accumulation of reels of recorded magnetic tape. This profile is derived directly from the fifth line on the data recording and transmission profile.

Figures 2-13 and 2-14 are the tape storage profiles for Mission 2, from which it is evident that no daily accumulation of tape occurs on either flight.

2.2.4.4 Data Collection Summary

The data collection requirements for Mission 2 are summarized in Tables 2-10 and 2-11 for Flights A and B, respectively.

The first column designates each experiment by experiment number (see 38, 39, 40, and 77). The exception to this is series 0100, the biomedical experiments, and series 0200, behavioral experiments. The second column is the maximum number of special analog channels that must be recorded at any one time because of that experiment. The third column indicates whether or not the digital output of the PCM multiplexer must be recorded. The fourth column indicates the minutes of recording time required by each experiment sequence. The fifth and sixth columns indicate the total number of sequences and total minutes of recording time respectively, required by that experiment during the course of the mission.

The next column contains a graphically-derived value of the total transmission time available during the mission and is entered only in the over-all totals. The figure given for "Recorded Backlog" is the difference between the over-all total "high rate" PCM and analog data recording time, and the over-all total available transmission time. Further experiment data collection details are given in Part II, Section 3. The tables for Mission 2 indicate that no excess magnetic tape needs to be recovered from either flight. The available estimates of film weight indicate that no problem exists in recovering the required amount of film. However, weight estimates for some of the film requirements were not available. Weight or volume estimates were also not available for several other forms of data.

2.2.5 TRAJECTORY

2.2.5.1 General

A ground trace of the early portions of the mission trajectory is shown in Figure 2-15. Vehicle A is launched at an azimuth of 83 degrees. The S-IB and S-IVB stages achieve an 80-n. mi. parking orbit inclined 29.5 degrees to the equatorial plane, in 9 minutes, 55 seconds (see 49 - Table 1). The nominal parking orbit time for Vehicle A is 3 hours, zero minutes, during which transposition and docking is performed along with other required crew tasks. The SPS is then ignited for Hohmann transfer to the 200-n. mi. mission orbit (19).

MAGNETIC TAPE STORAGE PROFILE
MISSION 2A

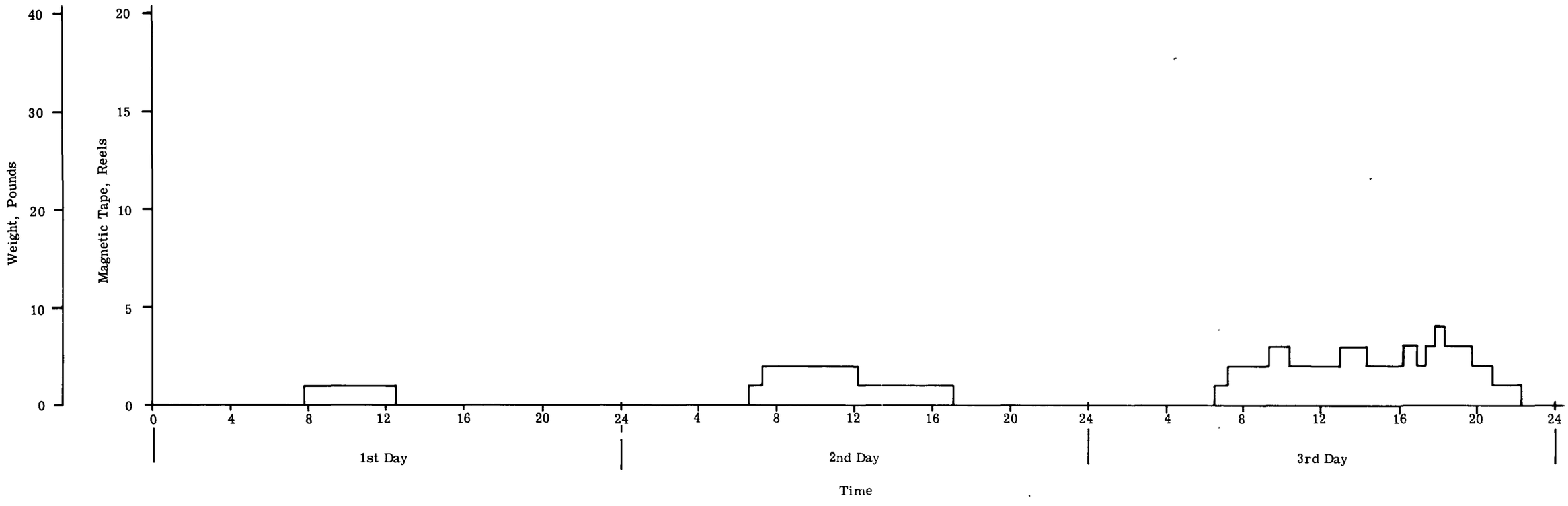


Figure 2-13. Magnetic Tape Storage, Mission 2, Launch A

2-83-1

2

MAGNETIC TAPE STORAGE PROFILE

MISSION 2 B

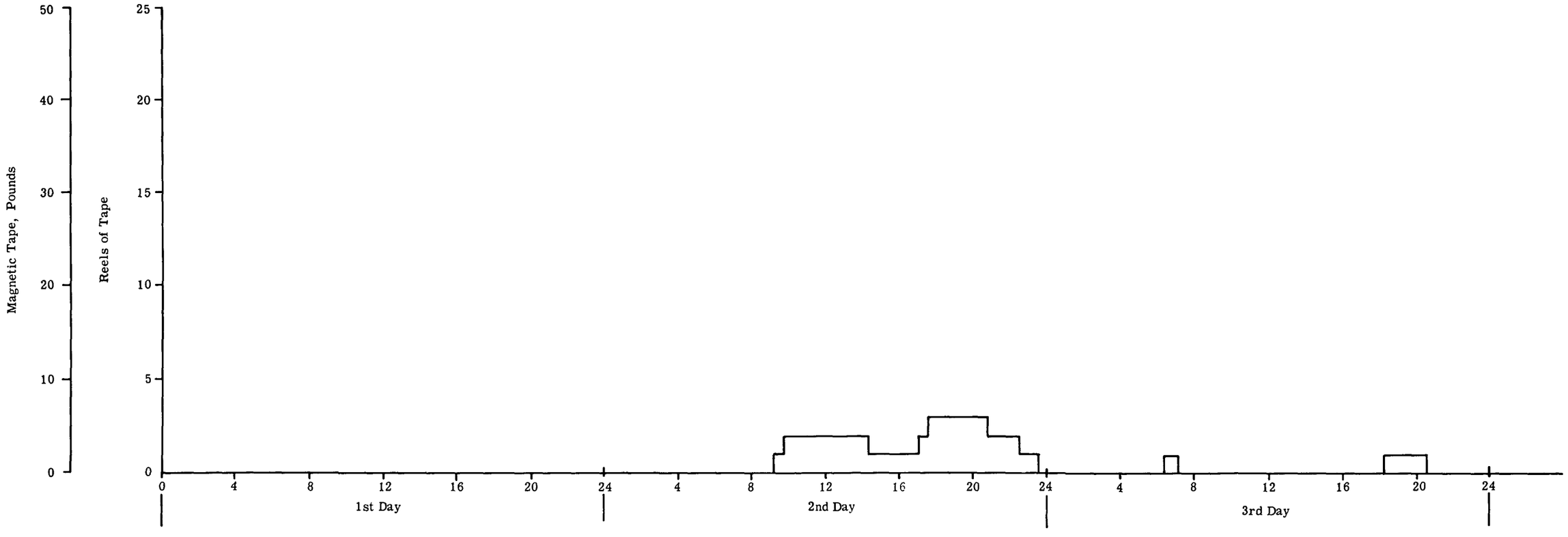


Figure 2-14. Magnetic Tape Storage, Mission 2, Launch B

Table 2-10

Data Collection Summary, Mission 2A

Experiment	Data Recording							Recovery Requirements				Remarks
	Max. No. of Channels in Use at One Time		Minutes per Sequence	Total No. of Sequences	Total Minutes Recorded	Transmission Time Avail. (Minutes)	Recorded Back Log (Minutes)	Magnetic Tape		Pounds of Film	Miscellaneous	
	Anal.	Dig.						Reels	Lbs.			
	*	*				**	***	**	**			* Assumed time sharing of channels as required. ** Summed only for total mission. ***Difference - Recorded minus Transmitted - only for total mission.
0100	6	1	*	119**	4342					X	X	* Varies for each experiment in series (see Data Collection Charts, Part II, Section 3). ** Varies for each experiment in series; number (119) is total for series for which recorder is used.
0200	1	1	*	18**	1410					145	X	* Varies for each experiment in series (see Data Collection Charts, Part II, Section 3). ** Varies for each experiment in series; number (18) is total for series for which recorder is used. X Indicates a requirement exists but no information is available.

Table 2-10

Data Collection Summary, Mission 2A (Cont.)

Experiment	Data Recording							Recovery Requirements				Remarks
	Max. No. of Channels in Use at One Time		Minutes per Sequence	Total No. of Sequences	Total Minutes Recorded	Transmission Time Avail. (Minutes)	Recorded Back Log (Minutes)	Magnetic Tape		Pounds of Film	Miscellaneous	
	Anal.	Dig.						Reels	Lbs.			
0501		1	2	40	80						X	
0601		1	330	1	330					7*	X**	* 2800 feet of 16-mm film. **One picture for each static test (10 total)
0602A	1	1	30	5	150					10		
B										X*		* 4000 feet of 16-mm film.
0603A		1	55	10	550					X*		* 200 to 400 feet of film per active orbit.
B			30	3	90					10		
0604		1	40	2	80					X		
0605			70	20	1400					X	X*	* 2 pounds of samples plus 10 photographs.
0606			**	1	**						30*	* Photo emulsion package. **10 to 14 days. X Indicates a requirement exists but no information is available.

Table 2-10

Data Collection Summary, Mission 2A (Cont.)

Experiment	Data Recording							Recovery Requirements				Remarks
	Max. No. of Channels in Use at One Time		Minutes per Sequence	Total No. of Sequences	Total Minutes Recorded	Transmission Time Avail. (Minutes)	Recorded Back Log (Minutes)	Magnetic Tape		Pounds of Film	Miscellaneous	
	Anal.	Dig.						Reels	Lbs.			
1501	1	1	65	6	390					X*	X**	* 200 feet at 16 frames per second **150 pictures.
1502	1	1	139*	17	2368					X	X	* Average minutes per sequence.
1503		1	123	12	1476					12*		* 6 rolls of film per flight; still and motion pictures - 12 pounds.
1504	1	1	340	2	680					X*		* 5 rolls of 8- or 16-mm film.
1505	1		10	36	360						X	X Indicates a requirement exists but no information is available.
Total	6**	2**			11256*	15967	0	0	0	184+	32+	* Time for 0601, 0603, 0604 not included in that data is low rate recording; 0605 and 0606 data not collected on magnetic tape. **Total number of channels in use at one time during mission, not a column summation; assumes time sharing as required.

Table 2-11

Data Collection Summary, Mission 2B

Experiment	Data Recording							Recovery Requirements				Remarks
	Max. No. of Channels in Use at One Time		Minutes per Sequence	Total No. of Sequences	Total Minutes Recorded	Transmission Time Avail. (Minutes)	Recorded Back Log (Minutes)	Magnetic Tape		Pounds of Film	Miscellaneous	
	Anal.	Dig.						Reels	Lbs.			
	*	*				**	***	**	**			* Assumed time sharing of channels as required. ** Summed only for total mission. ***Difference - Recorded minus Transmitted - only for total mission.
0100	6	1	*	118**	3978					X	X	* Varies for each experiment in series (see Data Collection Charts, Part II, Section 3). ** Varies for each experiment in series; number (118) is total for series for which recorder is used.
0501		1	2	80	160					X		X Indicates a requirement exists but no information is available.
0302	2	1	15	132	1980							

Table 2-11

Data Collection Summary, Mission 2B (Cont.)

Experiment	Data Recording							Recovery Requirements				Remarks
	Max. No. of Channels in Use at One Time		Minutes per Sequence	Total No. of Sequences	Total Minutes Recorded	Transmission Time Avail. (Minutes)	Recorded Back Log (Minutes)	Magnetic Tape		Pounds of Film	Miscellaneous	
	Anal.	Dig.						Reels	Lbs.			
0401			7	6	42					X	X	
0402			4	40	160					X	X	
0403			28	45	1260							
0404			8	45	360					X	X	
0405												
0406			10	6	60							
Total	6***	1***			6118*	15967	0	0	0	X**	X**	* Time for 0400 series not included because the data is not collected on magnetic tape. ** Insufficient information to determine weight. ***Total number of channels in use at one time during mission, not a column summation; assumes time sharing as required X Indicates a requirement exists but no information is available.

After coasting in the transfer orbit for 45 minutes, the SPS is reignited and insertion into mission orbit is established (15:55 GMT). During the 45-day mission the orbit will decay significantly and orbit restoration maneuvers will be performed three or four times during the mission.

Vehicle B is launched nominally 44 days later at a launch time and azimuth within the defined window (see Part II, Section 5). The launch azimuth range is 81.5 to 98.5 degrees for an assumed 24 minute launch window. The launch sequence for the 80-n. mi. parking orbit is the same as that for Vehicle A. When the proper phasing is achieved after a wait in the parking orbit, Vehicle B performs the insertion to a Hohmann transfer orbit, and coasts to apogee for insertion to the mission orbit at 200 n. mi. Final rendezvous maneuvers are performed and a hard dock is completed. After the necessary tasks are performed, the vehicles separate and Vehicle A performs the deboost maneuver. After deboost, the service module is jettisoned and the command module re-enters the atmosphere. Forty-five days later, Vehicle B performs the same sequence and re-enters.

2.2.5.2 Assumptions

The following assumptions were used to generate the ground trace plots for the first vehicle (Figure 2-11).

a. Altitude = 80 n. mi.
b. Latitude = 28.11 degrees
c. Longitude = minus 62.61 degrees
d. Relative azimuth = 100.2 degrees
e. Launch time = 1200 GMT

2.2.5.3 Orbit Decay

Orbit decay is a significant factor in stay times. The 1962 ARDC model atmosphere yields a 7-n. mi. decay in a 200-n. mi. orbit over 5 days, in contrast to the pessimistic model of Reference 2, which yields a decay to 165 n. mi. The SPS fuel requirements for the 7-n. mi. decay is 77 pounds; 300 pounds (using 3-orbit restoration spaced 15 days apart) is required to compensate for the consequences of the pessimistic model. A ballistic coefficient of 0.1445 feet^2/slug was used. Decay at the 80-n. mi. parking orbit is 1.1 n. mi. per orbit (ARDC 1962 atmosphere). Twelve pounds of fuel per orbit are required for orbit maintenance. (See Volume 1 Part II.)

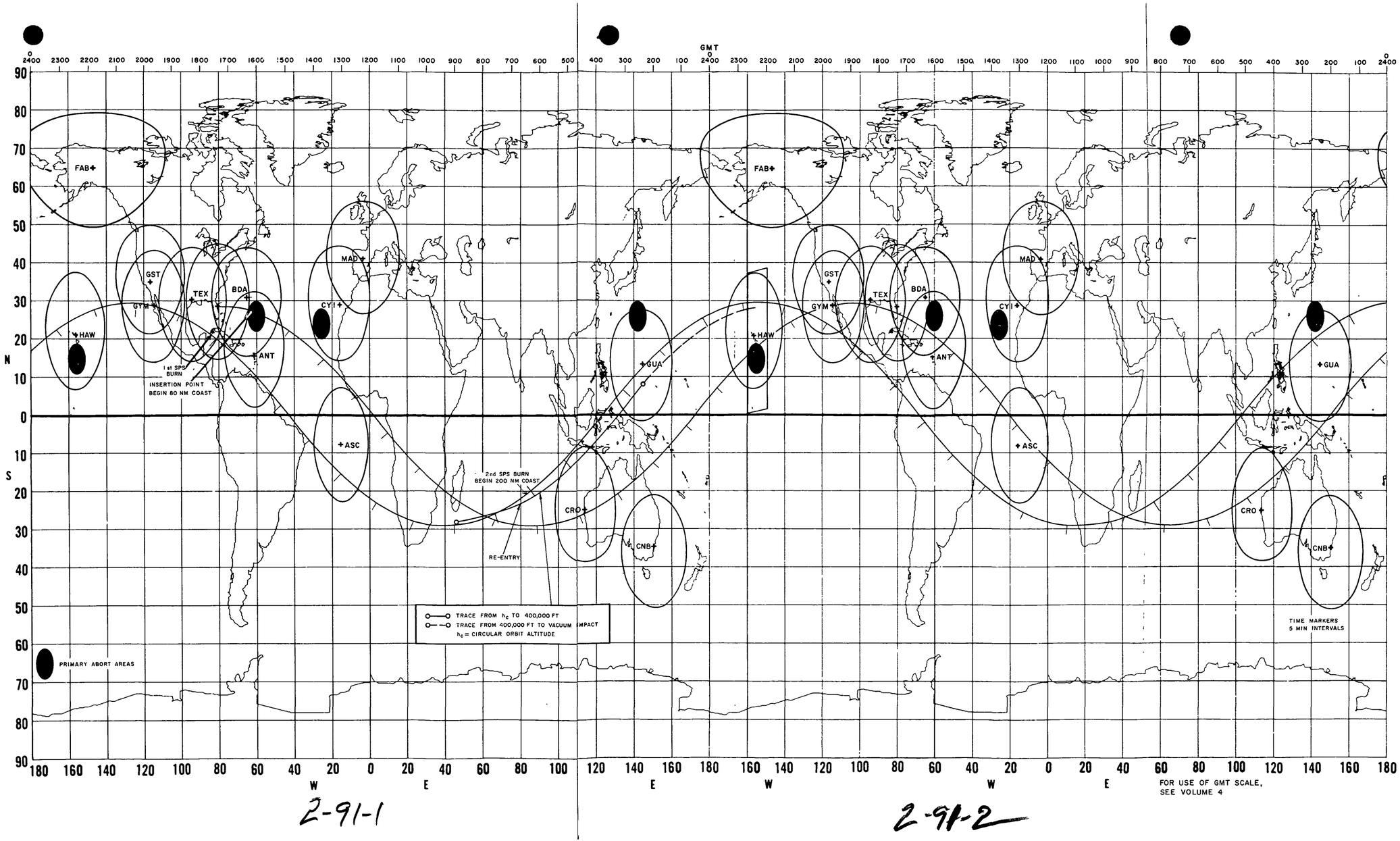


Figure 2-15. Mission 2, Vehicle A Launch, Mission Orbit Insertion and Re-entry

2.2.5.4 Rendezvous

In the selection of a launch window for Vehicle B, two considerations must be made:

a. The plane change capability of both vehicles.

b. The phasing between the vehicles at the time of Vehicle B launch.

Each of these constraints produces individual launch windows, and a "true" launch window is determined from a superposition of these two windows.

After consideration of payload capabilities, the plane change launch window was selected to be 24 minutes, corresponding to a plane change capability of 0.32 degrees (see 50) and a resulting SPS fuel requirement of 455 pounds, distributed between the two vehicles. Using these values, a 10-day period provides five launch opportunities, two of which will occur on consecutive days.

In the example shown on the geographic map, Figure 2-16, a 9.5-minute launch delay which corresponds to a 2-hour phasing time (4 hours, 52 minutes parking orbit time) is allowed for Vehicle B. At the start of rendezvous, Guam and Hawaii are available for tracking. Hard docking should be completed while tracking coverage is still available from Hawaii.

Experiment constraints on trajectory-related parameters for this mission are the following:

a. First Vehicle

(1) Seasonal requirements: None.

(2) Illumination requirements: None.

(3) Vehicle orientation: Earth pointing (35).

(4) Stabilization requirements: ±5 degrees (Normal), ±0.5 degree (19 hours) (35).

(5) Ephemeris accuracy requirements: Not specified.

b. Second Vehicle

(1) Seasonal requirements: None.

(2) Illumination requirements: None.

(3) Vehicle orientation: Various (35).

(4) Stabilization requirements: ±5 degrees (4 hours), ±5 degrees (Normal) (35).

(5) Ephemeris accuracy requirements: Not specified.

2.2.5.5 Flight Parameters, Ground Coverage, and Illumination

A detailed listing of flight dynamic parameters for all powered phases of the mission is given in Table 2-12.

Data for the launch from liftoff through injection into the 80-n. mi. circular orbit is taken from Reference 49. Computation by General Electric has supplied the remainder of the data. Ground tracking coverage and illumination for the ascent phases is given in Figure 2-17.

The ground traces for the later orbits can be obtained by using the uniform grid projection with ascending nodes plotted on it in conjunction with transparent overlays. These materials are included in Volume IV of this report.

2.2.6 ABORT ANALYSIS

2.2.6.1 General

Four major abort modes are applicable to Mission 2. They are:

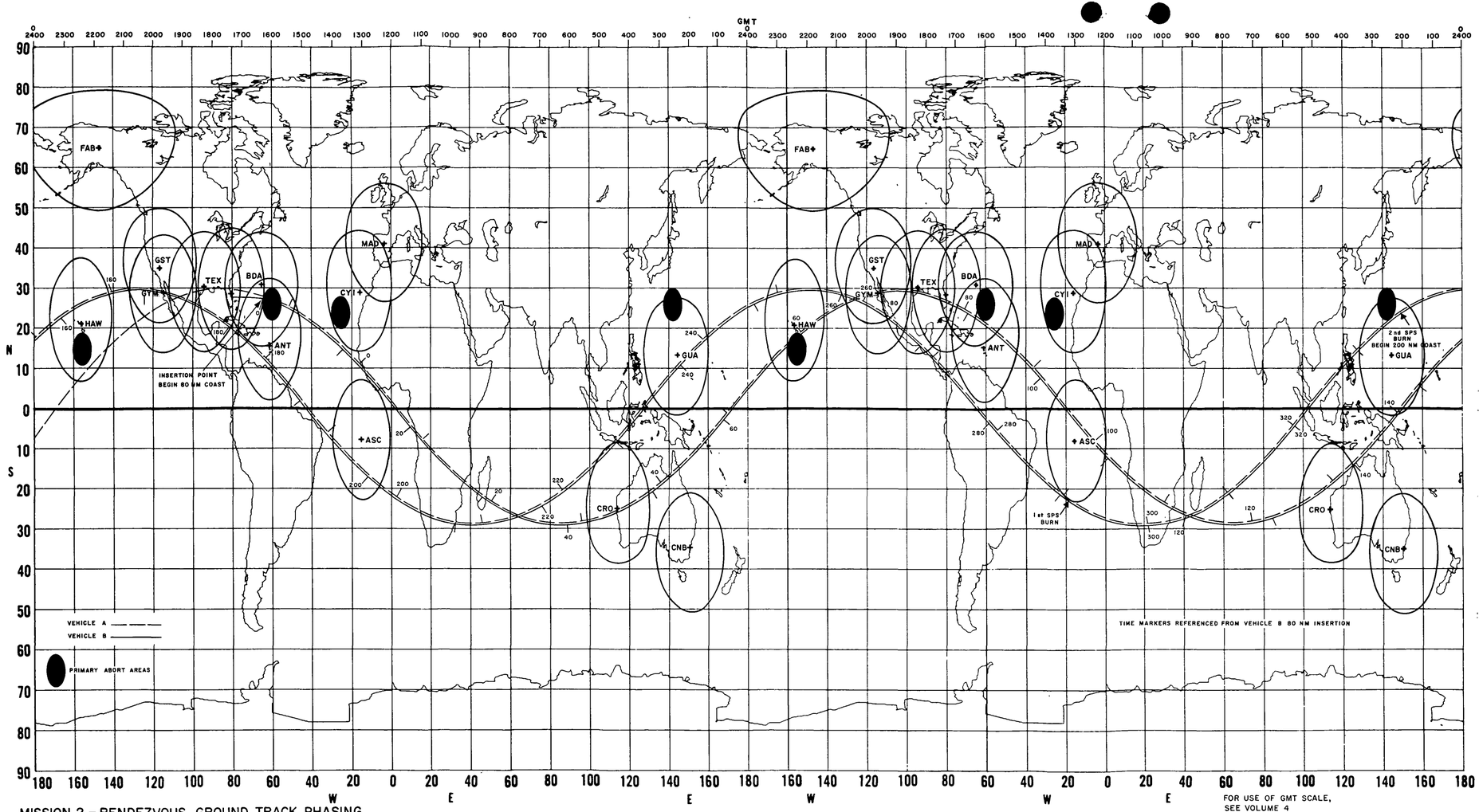
a. LES Abort.
b. Suborbital Abort.
c. Abort-to-orbit.
d. De-orbit Abort.

The relationship between the mission phases and the abort mode available at any particular point within the phase, for both flights of the mission, is shown in Figure 2-18. For each flight, ground rules for each of these major abort modes are defined, and abort mode descriptions and logic diagrams for each abort mode are presented.

2.2.6.2 Launch Escape System

2.2.6.2.1 General

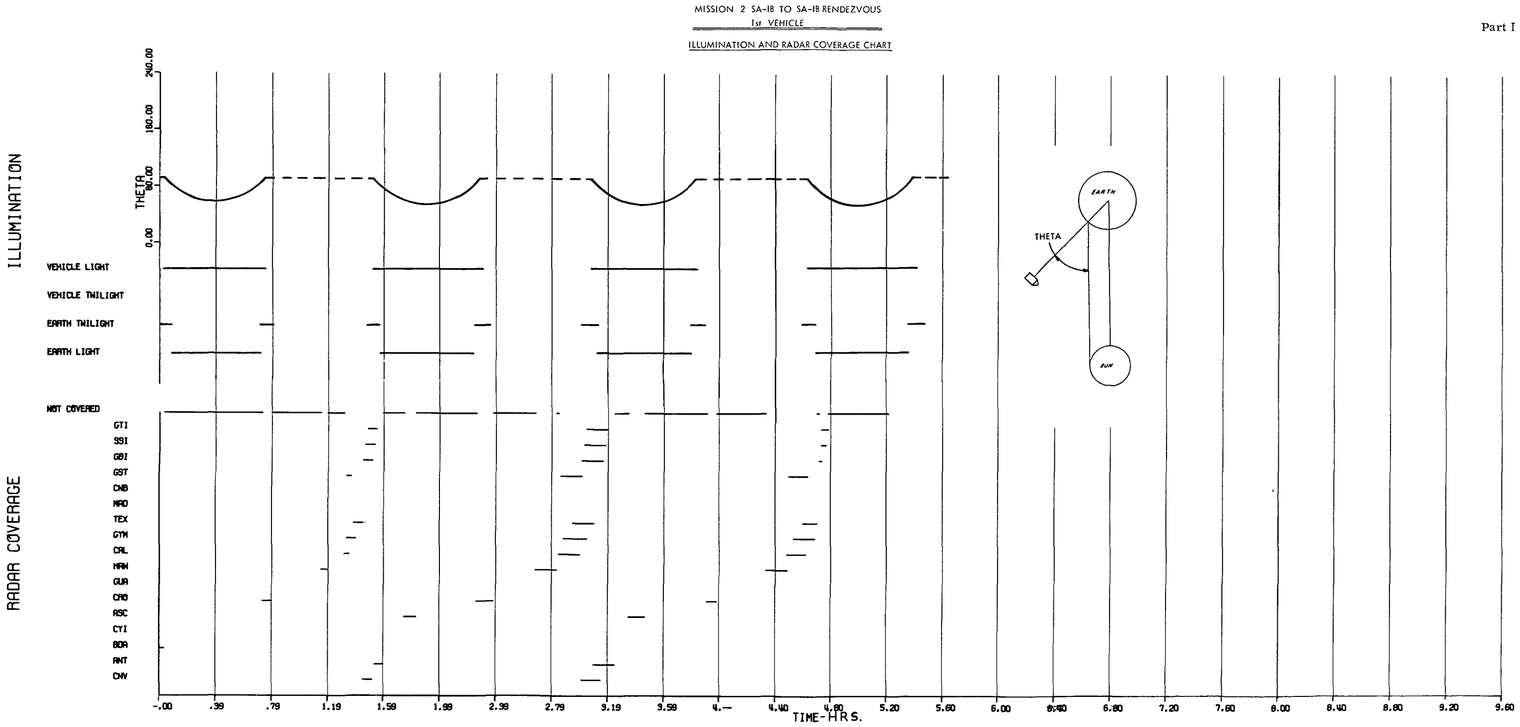
A launch escape system (LES) abort may be performed on Flight A, and on Flight B any time between LES enablement during prelaunch and LES jettison during the second stage thrusting period. The LES ground rules, mode descriptions, and logic diagrams are the same for both flights of Mission 2.



MISSION 2 – RENDEZVOUS GROUND TRACK PHASING

Figure 2-16. Rendezvous of Vehicles B and A

Part I



Mission 2 Saturn IB to Saturn IB Rendezvous (Coverage and Illumination)

Figure 2-17. Illumination and Radar Coverage

2-96-1

2-96-2

Table 2-12

Mission 2 - Saturn IB to Saturn IB Rendezvous
Saturn IB Reference Trajectories; 200-N. Mi. Circular
Orbit by Hohmann Transfer (Perigee = 80 N. Mi., i = 26.5 Degrees)[10]

Trajectory Event	Time (seconds)	Geodetic Latitude (degrees)	Longitude (degrees)	Velocity[1] (feet per second)	Altitude (feet)	Flight Path Angle[1] (degrees)	Azimuth[2] (degrees)	Vehicle Weight (pounds)
1. Liftoff	0.00	28.65	-80.64	1,340	0	0.00		1,298,819
2. End Vertical Rise, Begin Zero-Lift Flight	10.00	28.65	-80.64	1,345	506	4.50	90.0	1,234,545
3. Shutdown of S-IB Inboard Engines	134.72	28.64	-80.11	7,242	176,584	27.36	90.5	434,345
4. Shutdown of S-IB Outboard Engines (Burnout) Begin Coast	141.64	28.64	-80.00	7,623	199,816	26.34	90.6	412,472
5. Jettison of S-IB; S-IVB Ignition: Mixture Ratio = 5.0; Constant Attitude Rate	145.94	28.64	-79.93	7,564	214,082	25.50	90.6	306,646[3]
6. Jettison Ullage Cases and Thermolag, Change Mixture Ratio to 5.5	155.94	28.64	-79.75	7,648	245,726	23.69	90.8	301,599[4]
7. Jettison Launch Escape System	165.94	28.64	-79.58	7,772	275,618	21.96	90.9	287,963[5]
8. Change Mixture Ratio to 4.7	440.00	28.22	-71.48	16,341	557,566	- 1.18	95.7	138,981
9. S-IVB Burnout (80-n. mi. Parking Orbit) End Integrated Trajectory	595.54	27.11	-62.61	25,665	486,007	0.00	100.2	69,755
10. Service Module Ignition for Perigee Maneuver	3:10:00[9]	21.00	-84.10	25,665	80 N. Mi.	0.00		34,770[6]
11. Service Module Burnout, Coast to Apogee[7]	3:10:12	21.00	-84.10	25,879	80 N. Mi.	0.00		33,968
12. Service Module Ignition for Apogee Maneuver	3:55:00	-21.00	83.00	25,016	200 N. Mi.	0.00		33,968

Table 2-12

Mission 2 - Saturn IB to Saturn IB Rendezvous
Saturn IB Reference Trajectories; 200-N. Mi. Circular
Orbit by Hohmann Transfer (Perigee = 80 N. Mi., i = 26.5 Degrees)[10] (Cont.)

Trajectory Event	Time (seconds)	Geodetic Latitude (degrees)	Longitude (degrees)	Velocity[1] (feet per second)	Altitude (feet)	Flight Path Angle[1] (degrees)	Azimuth[2] (degrees)	Vehicle Weight (pounds)
13. Service Module Burnout	3:55:11	-21.00	83.00	25,229	200 N. Mi.	0.00		33,185[10]
14. Service Module Ignition for Deboost Maneuver	00:00:00				200 N. Mi.	0.00		Mission A 21,057[8] Mission B 21,421
15. Service Module Burnout[11]	00:00:12	-28.50	47.00		200 N. Mi.	0.00		Mission A 20,247 Mission B 20,247
16. Command Module Re-Entry	00:26:47	+ 8.00	-145.00		66 N. Mi.	-1.52		Mission A 10,303 Mission B 10,303

(1) Inertial Quantities.
(2) Relative Quantities.
(3) After Jettison Event of 105,826 Pounds.
(4) After Jettison Event of 235 Pounds.
(5) After Jettison Event of 8,200 Pounds.
(6) After Jettison Event of 34,985 Pounds.
(7) Each S/M operation provides an additional 10-percent velocity contingency over the velocity increment required.
(8) The difference between the weights of events 13 and 14 represents payload left in orbit or expended during stay.
(9) Allows three hours for crew tasks (Hours: Minutes: Seconds).
(10) For inclination of 29.5 degrees, reduction in weight is 90 pounds (Reference 49, Table 1).
(11) Assumes re-entry immediately after injection (see Part II, Section 6).

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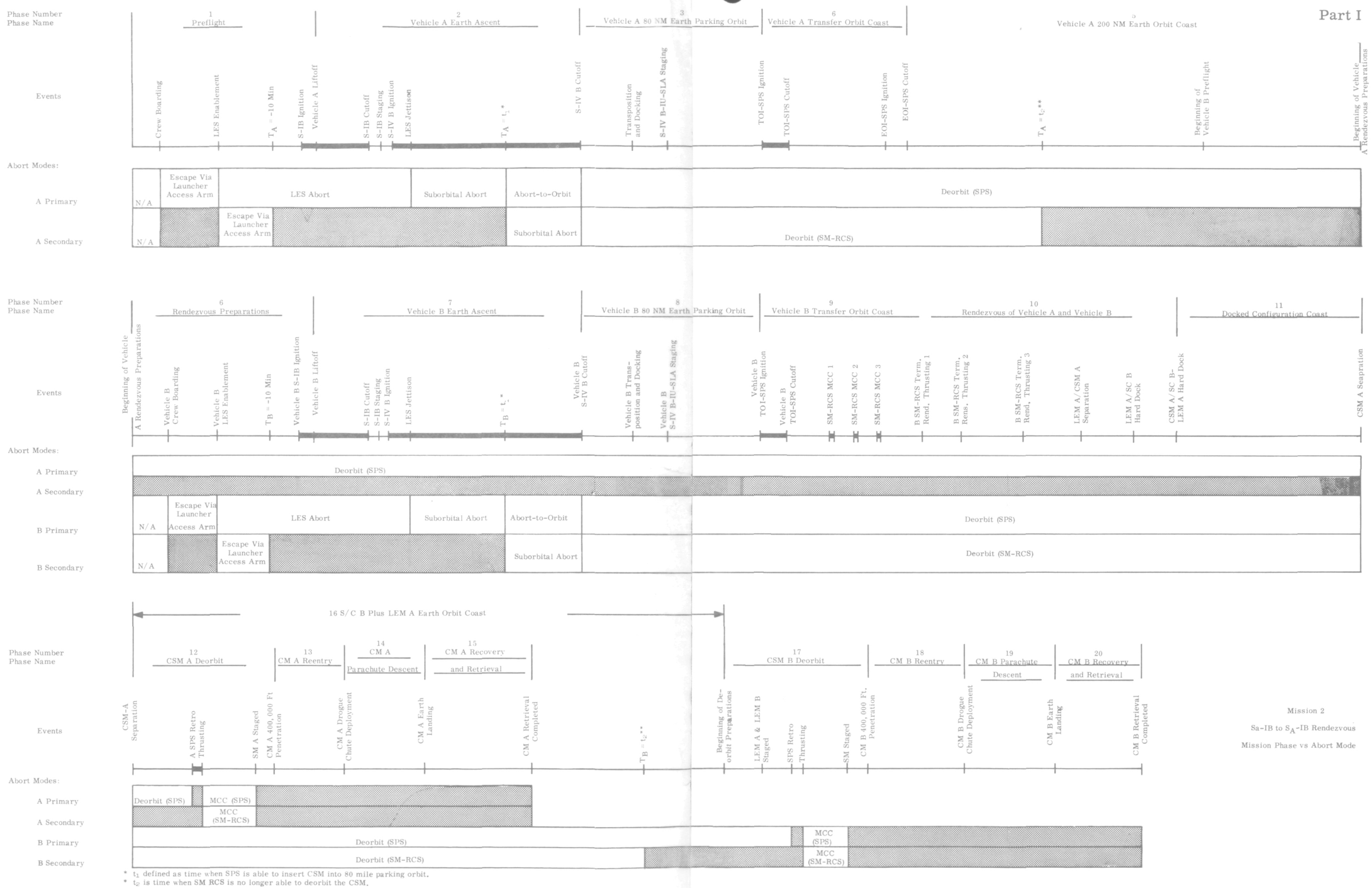


Mission 2

Sa-IB to S_A-IB Rendezvous

Mission Phase vs Abort Mode

* t_1 defined as time when SPS is able to insert CSM into 80 mile parking orbit.

* t_2 is time when SM RCS is no longer able to deorbit the CSM.

Figure 2-18. Mission Phase versus Abort Mode

2-99

2.2.6.2.2 Ground Rules

The (LES) abort is the primary abort mode from the time of LES enablement during the prelaunch terminal countdown to LES jettison during the Saturn IVB stage thrusting period.

Launch vehicle engine shutdown must precede LES ignition for all aborts except those initiated during the first 40 seconds of flight. During the first 40 seconds, engine shutdown is inhibited by an AMR range safety constraint.

An automatic sequencing of spacecraft events is provided for aborts initiated below an altitude of 120,000 feet. Above 120,000 feet, the astronauts are required to jettison the LES manually, after escape motor thrust tail-off, and to orient the spacecraft to a heat-shield-forward attitude until reaching 12,000 feet on descent, when the switch is closed again.

If the abort occurs above an altitude of approximately 120,000 feet, the canard subsystem is ineffective, and the pilot must manually orient the CM in a heat-shield-forward attitude. To do so, the LES must be manually jettisoned by the pilot after launch escape motor thrust tail-off (about 8 seconds). The CM apex cover is left on. Automatic RCS rate damping begins at tower jettison. Upon reaching 25,000 feet on descent, the apex cover is jettisoned upon closing of the previously discussed barometric switch. Two seconds later the drogue parachutes are deployed. At 12,000 feet another barometric switch closes, and disconnects the drogue chutes to deploy the main chutes.

2.2.6.2.3 Mode Description

The operational events that follow the initiation of a LES abort, controlled by sequential timers and barometric switches, are shown in Figure 2-19. At abort initiation, launch vehicle engine shutdown is commanded (except during the first 40 seconds of powered flight when shutdown is inhibited by a range safety lockout). The launch escape and pitch control motors are then fired simultaneously. The pitch motor burns for approximately 0.5 second to provide lateral separation from the booster, while the escape motor thrusts for about 3.5 seconds to provide longitudinal separation.

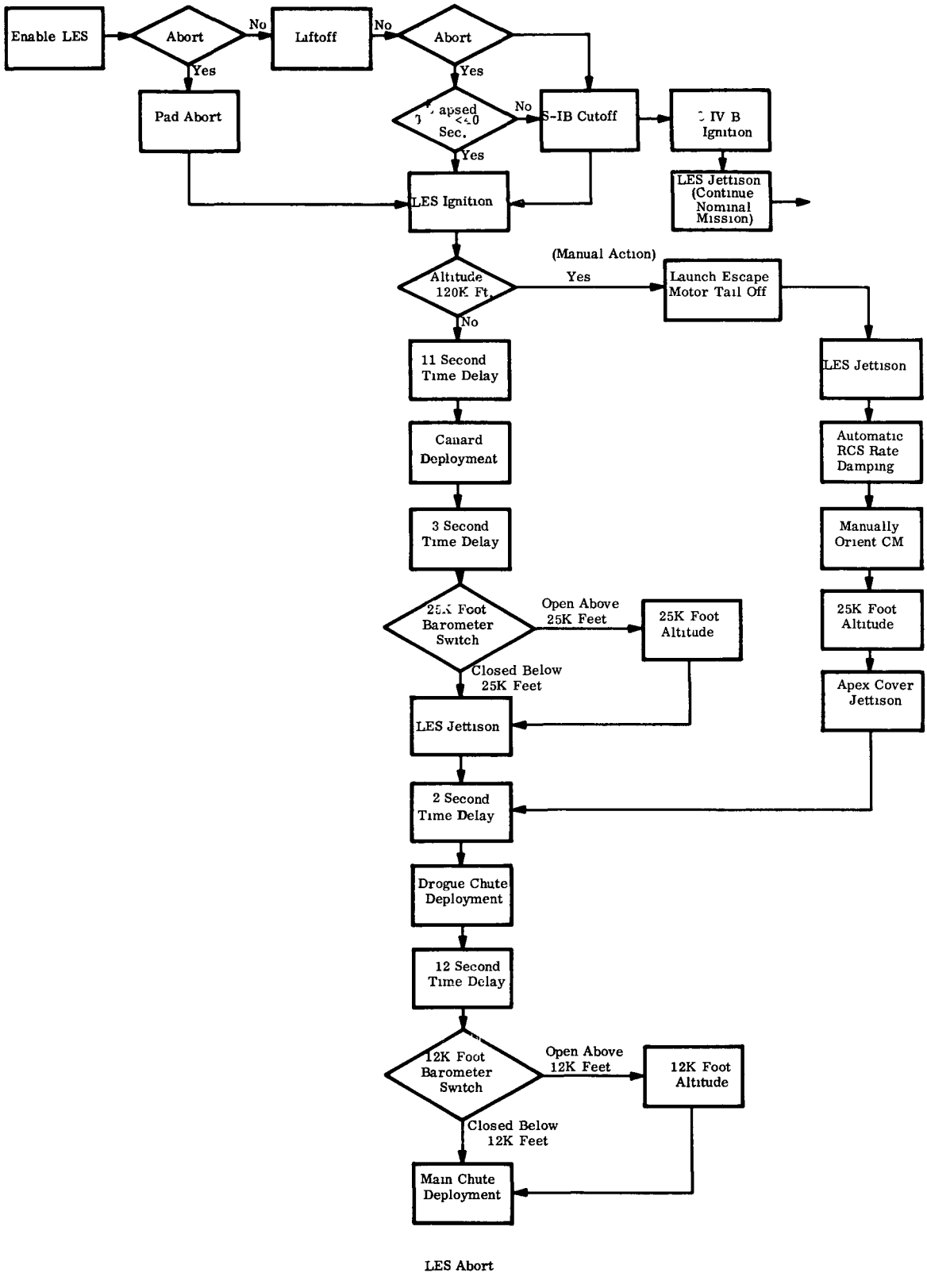


Figure 2-19. LES Abort

At 11 seconds after abort, the canards are deployed. At 14 seconds after abort, if a 25,000-foot barometric switch (closed during prelaunch) is still closed, the tower jettison motor separates the LES, including the CM apex cover, from the CM. If the switch is open, LES jettison is delayed until reaching 25,000 feet, during descent, when the barometric switch is closed again. Drogue parachute deployment occurs two seconds after LES jettison. Main parachute deployment is controlled by a timer and barometric switch similar to that of LES jettison. At 12 seconds after drogue deployment, if a 12,000-foot barometric switch is still closed, the drogue chutes are disconnected and the main chutes are deployed. If the switch is open, the drogue chutes are retained.

2.2.6.3 Suborbital Abort

2.2.6.3.1 General

A suborbital abort may be performed by Flight A or Flight B any time between LES jettison and earth parking orbit insertion. The suborbital abort ground rules, mode descriptions, and logic flow diagrams are identical for either flight of the mission.

2.2.6.3.2 Ground Rules

A suborbital abort is the primary abort mode from LES jettison to t_1 seconds after liftoff. A suborbital abort is the secondary abort mode from t_1 seconds after liftoff to S-IVB cutoff at the completion of the earth parking orbit insertion maneuver. A suborbital abort in the primary mode results in a CM landing in the AMR. In the secondary mode, the CM lands in the Indian Ocean recovery area. The primary abort mode requires that the CM fly a maximum lift trajectory during entry to avoid excessive decelerations. The secondary abort mode requires SPS posigrade thrust to attain the range of the landing site. A constant bank angle of approximately 60 degrees will be flown by the CM during entry. Launch vehicle shutdown must precede the initiation of an abort action.

The CSM abort separation procedure will utilize both an SM RCS and SPS burn of fixed initiation time, ΔV, and attitude. The SM must be separated from the CM at some fixed increment of time, after SM engine shutdown following completion of the suborbital abort maneuvers.

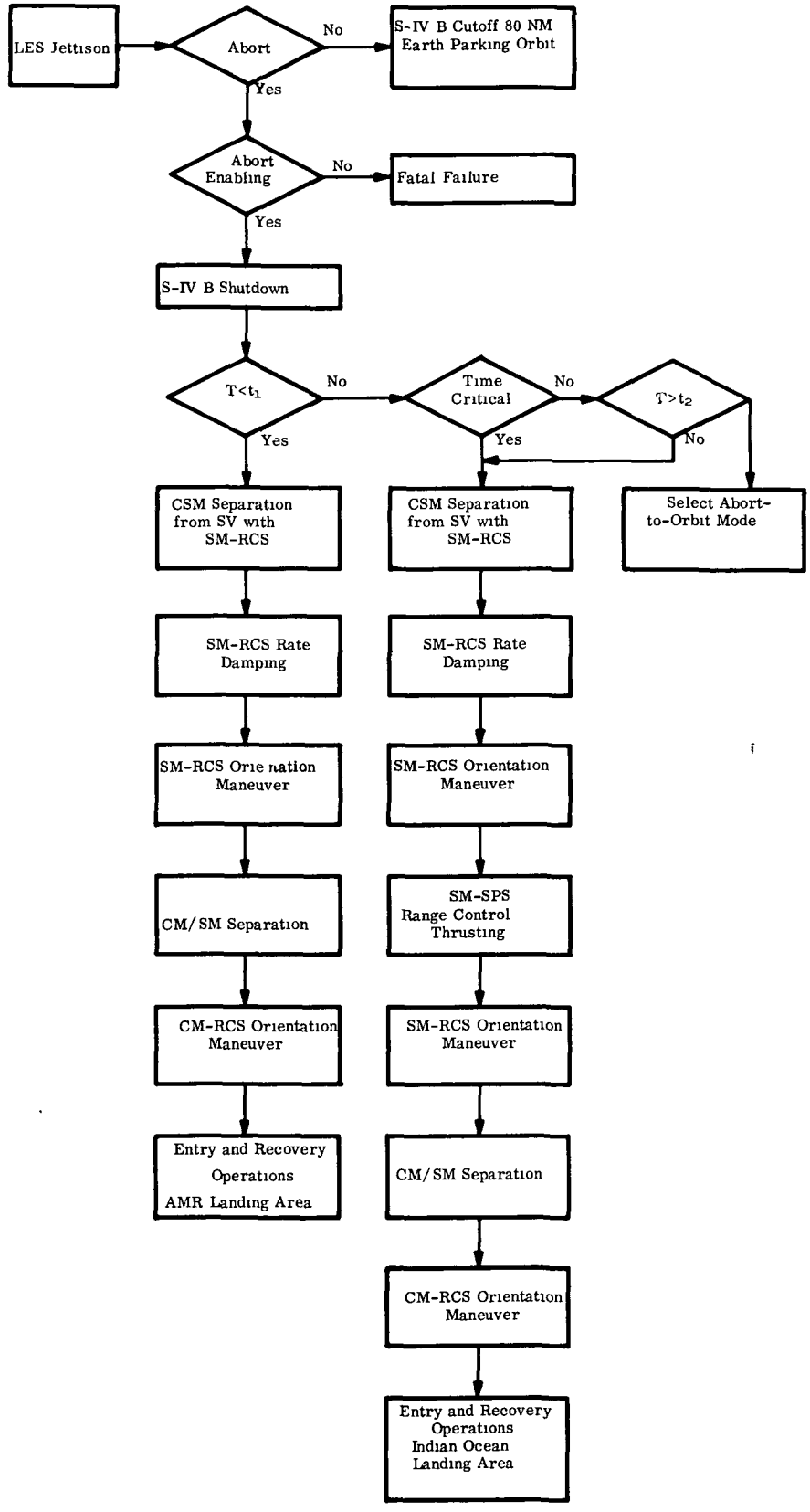
2.2.6.3.3 Mode Description

A detailed sequence of events in a suborbital abort is shown in Figure 2-20. Following booster shutdown, the SM RCS is used to provide approximately 45 inches of separation between the CSM and launch vehicle to clear the SPS engine nozzle of the spacecraft LEM adapter (SLA). The SPS engine nozzle must be clear of the SLA before the SPS can be ignited. It may be necessary to fire the SPS as soon as possible to damp out any existing pitch and yaw rates, which could exist at abort due to a launch vehicle control system failure. After the SM RCS separation and SPS damping maneuvers are completed, a turnaround to the required thrust attitude is achieved by pulsing the SM RCS.

Depending on the range control required, the SPS can be burned for variable times in either a posigrade or retrograde attitude. The time delay to thrust initiation can also be varied. After the CM is separated from the SM, the CM is capable of re-entering ballistically or with variable lift.

SPS range control thrusting is not used for aborts initiated during the period between LES jettison and t_1 seconds after liftoff. A suborbital abort is the primary and only available abort mode during this period. Range control thrusting for aborts during this period is neither desired nor required, since the CM flying a maximum lift trajectory will land within the confines of the Atlantic Missile Range. Therefore, the primary abort mode procedure is to use the SM RCS to establish the correct CM/SM separation attitude after the SM RCS launch vehicle separation and SPS damping maneuvers are completed. The SM RCS must provide the CM/SM separation velocity as the CM has no translatory thrust capability. Following separation, the CM RCS is used to orient the CM for a maximum lift entry.

During the period from t_1 seconds, after liftoff to S-IVB cutoff at the completion of the earth parking orbit insertion maneuver, the suborbital abort mode is the secondary abort mode. SPS range control thrusting will be utilized for the suborbital abort mode during this period to avoid a possible landing on the African continent. SPS posigrade thrusting will be used to insure a CM landing in the Indian Ocean recovery area. Following SPS range control thrusting, the SM RCS is used to establish the correct CM-SM separation attitude, and then to provide the separation velocity. Following CM/SM separation, the CM RCS is used to orient the CM for a constant bank angle utilizing approximately half-lift capability.



Suborbital Abort Vehicle A

Figure 2-20. Suborbital Abort

2.2.6.4 Abort to Orbit

2.2.6.4.1 General

An abort-to-orbit is the primary abort mode for either spacecraft of Mission 2 from $T = t_1$, the time at which the SPS first gains the capability of inserting the CSM into the 80-n. mi. earth parking orbit, to nominal S-IVB cutoff, at the completion of the parking orbit insertion maneuver. All time critical aborts occurring during this period require the selection of the suborbital abort mode, the designated secondary abort mode for this time period.

The advisability of aborting to orbit once the capability exists depends on the malfunction. If crew safety is not diminished, one advantage of aborting to orbit is that more flexibility in landing-site selection is available.

2.2.6.4.2 Ground Rules

An abort-to-orbit is the primary abort mode from t_1 seconds after liftoff to S-IVB cutoff at parking orbit insertion. All time-critical aborts will be suborbital.

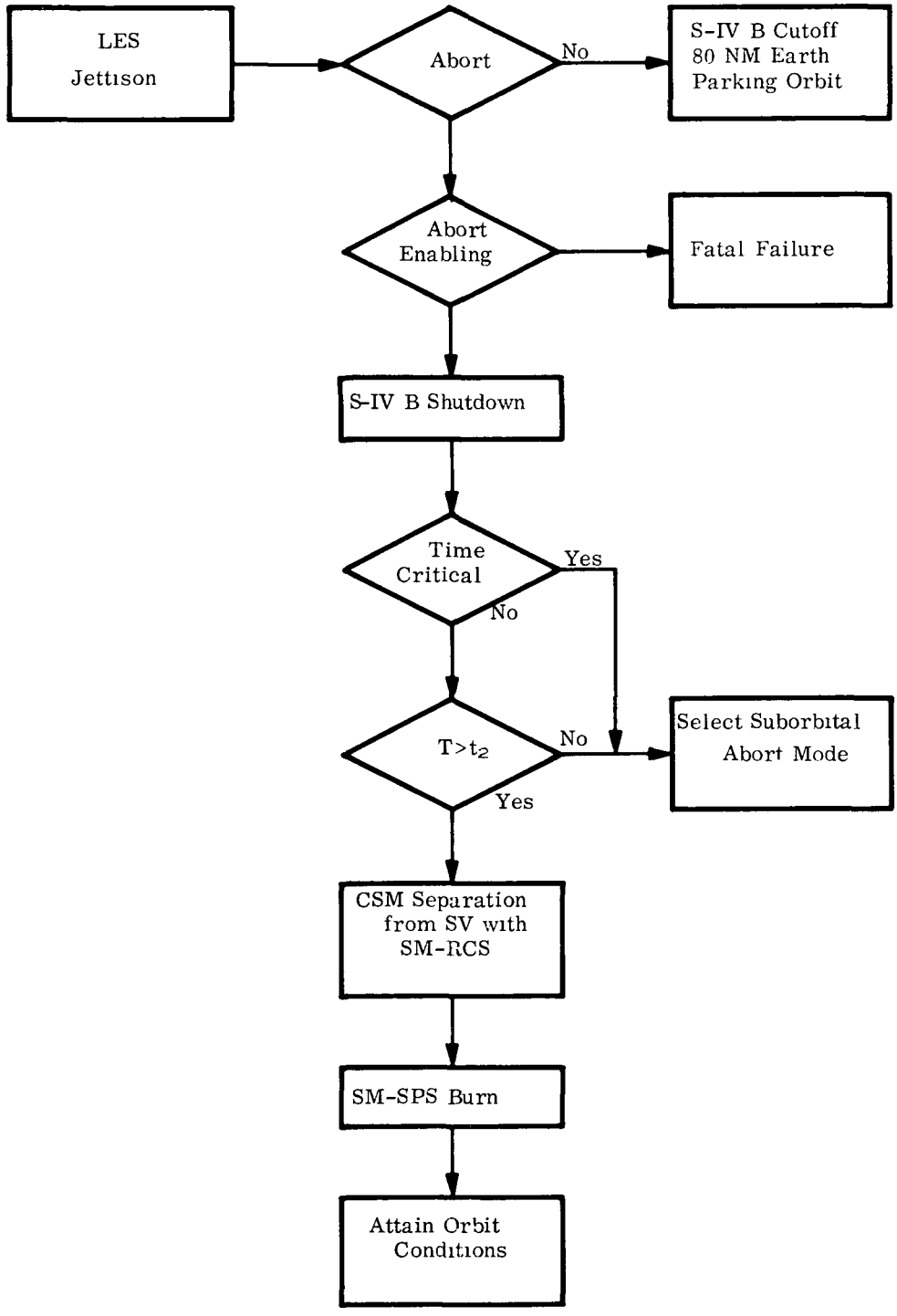
S-IVB shutdown shall precede initiation of spacecraft abort action. The CSM separation procedure for an abort-to-orbit will utilize both a SM RCS and SPS burn of fixed initiation time, ΔV, and attitude.

2.2.6.4.3 Mode Description

S-IVB shutdown must precede the initiation of spacecraft abort action. Following booster shutdown, the SM RCS must be used to provide approximately 45 inches of separation between the CSM and launch vehicle to clear the SPS engine nozzle of the SLA, a requirement which must be met before the SPS can be ignited. Following CSM separation, the SPS is ignited and thrusted until orbit insertion is achieved.

Sufficient SPS propellant must be reserved for the de-orbit maneuver. An SPS abort-to-orbit from an abort occurring at $T = t_1$ requires approximately ( ) seconds of SPS thrusting.

The sequence of events during an abort-to-orbit is shown in Figure 2-21.



Abort-to-Orbit

Figure 2-21. Abort-to-Orbit

2.2.6.5 De-orbit Abort

2.2.6.5.1 General

A de-orbit abort of Spacecraft A or B can occur after S-IVB cutoff and prior to SPS ignition for the scheduled de-orbit maneuver. Ground rules for de-orbit aborts are identical for both spacecraft in Mission 2; mode descriptions and logic diagrams for the two spacecraft are provided in paragraph 2.2.6.4.2 and Figures 2-22 and 2-23.

2.2.6.5.2 Ground Rules

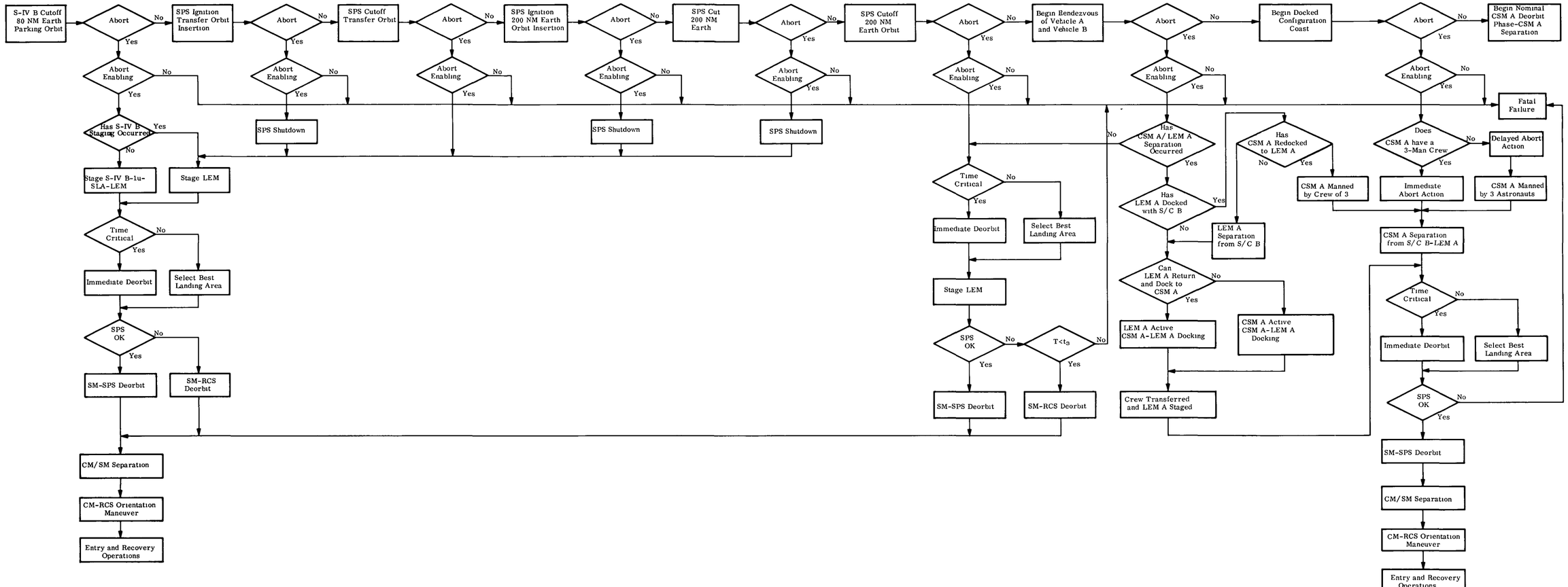
From S-IVB cutoff at earth parking orbit insertion and SPS ignition for the nominally scheduled de-orbit maneuver, the only abort possible is an early de-orbit maneuver. The primary abort mode utilizes the SPS to accomplish the de-orbit maneuver; the secondary abort mode uses the SM RCS. The secondary abort mode is not available after $T = t_3$. The de-orbit maneuver is executed by using a single propulsive burn at a fixed retrograde attitude and for a specified ΔV. The spacecraft shall consist of only the CSM, manned by the three-man crew, at the time of initiation of de-orbit thrusting. The CM/SM separation procedure will utilize an SM RCS translation and roll maneuver. When time permits, the de-orbit maneuver will be initiated at a time that permits entry to one of the planned recovery areas, preferably the primary recovery area.

2.2.6.5.3 Mode Description

If a de-orbit abort is required before the completion of transposition and docking, the S-IVB-IU-SLA-LEM configuration is staged; whereas, if transposition and docking is completed, only the LEM must be staged. Following staging operations, the CSM is oriented for the de-orbit maneuver using the SM RCS.

The SPS is the primary propulsive mode for the de-orbit abort. The SM RCS serves as a secondary propulsive mode until a mission time of t_2, the time at which the SM RCS propellant usage has depleted the ΔV capability to the point where it is no longer sufficient for the required de-orbit thrusting. If the de-orbit abort is required during transfer orbit insertion, earth orbit insertion thrusting, or any SPS thrusting period, the SPS must be shut down.

Time-critical aborts require immediate initiation of the de-orbit maneuver with minimum consideration given to the landing area. When the situation is not time-critical, the de-orbit maneuver can be delayed, as necessary, to land in a designated recovery area.



2-109-1

2-109-2

Figure 2-22. De-orbit Abort, Vehicle A

-3

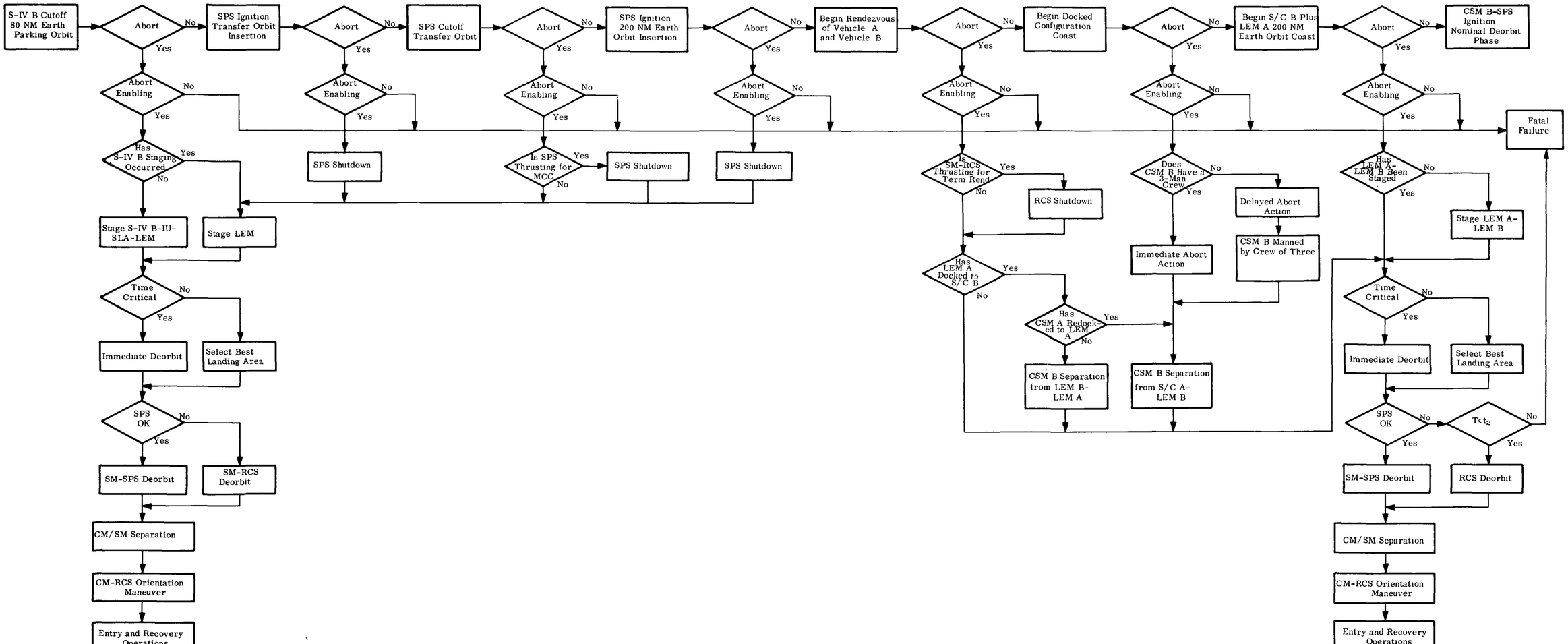


Figure 2-23. De-orbit Abort, Vehicle B

The sequence of events following the de-orbit abort thrusting maneuver is the same as for the events that follow the nominal de-orbit manuever.

If CSM A abort action is required during the period between CSM A/LEM A separation for rendezvous with Vehicle B and CSM A nominal redocking with LEM A, the abort procedure requires a CSM A/LEM A redocking, transfer of LEM A crew to CSM A, and LEM A/CSM A separation, before CSM A initiates de-orbit abort thrusting. Either the CSM A or LEM A can be the active vehicle in the redocking operation, but ordinarily the LEM A would be the active vehicle.

If CSM A abort action is required during the docking configuration coast, the abort procedure requires that CSM A be manned by a three-man crew before CSM A is separated from the Vehicle B/LEM A configuration. Following separation the de-orbit abort procedure is as described before.

If CSM B abort action is required during the period from LEM A docking to LEM B and nominal CSM A separation, the de-orbit abort procedure requires that the CSM B be manned by a three-man crew before CSM B is separated from the LEM A/LEM B or CSM A/LEM A/LEM B configuration. Following separation the de-orbit abort procedure is as described before.

The sequence of events during a de-orbit abort are shown in Figure 2-22 and 2-23 for Spacecraft A and Spacecraft B.

SECTION 3

MISSION 3

SYNCHRONOUS OBSERVATION POST

3.1 MISSION DESCRIPTION

3.1.1 GENERAL

Mission 3 establishes a synchronous observation post over three separate points of the earth's surface at an altitude of 19,350 n. mi., where experiments will be conducted in high resolution astronomical photography and associated calibrated photometry. The earth's polar auroras will also be studied, and studies will continue in the response of biological systems to the space environment. The duration of the mission is 45 days (see 19, 1).

3.1.2 MISSION SUMMARY

3.1.2.1 General

The spacecraft, with a crew of three astronauts, is launched with the Saturn V into an earth parking orbit at 100 n. mi., with a 28.5 degree inclination. All three Saturn V stages are used. Following normal checkout and orbit confirmation procedures, the S-IVB is utilized to place the spacecraft into the first transfer ellipse, which will have an apogee of 19,350 n. mi. Insertion takes place at a descending node of the 100-n. mi. parking orbit, and the S-IVB changes the plane of the first transfer ellipse 2 degrees with respect to the parking orbit. At apogee the S-IVB fires a third time to raise the perigee to about 5300 n. mi. A 17.4-degree plane change is made during the S-IVB third burn. Immediately after the third burn, transposition and docking occur and the S-IVB/IU/SLA configuration is jettisoned. The second transfer ellipse is then circularized at apogee, after one orbit, utilizing the SPS. A plane change of 9.1 degrees is accomplished during the SPS burn, placing the spacecraft in an orbit of 0-degree inclination at an altitude of 19,350 n. mi. This altitude yields an orbital duration of 24 hours, hence the subsatellite point is stationary with respect to the surface of the earth. In each transfer ellipse, the apogee and perigee lie at the nodes because the initial transfer was Hohmann.

Upon attaining synchronous orbit conditions, the experiment schedule is initiated and station-keeping activities begin. Station-keeping involves supplying incremental velocity vector in order to control the spacecraft period. The initial station is at 126 degrees west; the second station is at 70 degrees west, and the third station is at 90 degrees east (see 48).

Spacecraft station shifting is accomplished in the same manner over a period of days; that is, a drift rate will be established using the SPS system to place the spacecraft in an elliptical orbit, and the orbit will be circularized upon reaching the next station. At the termination of the experiment schedule, the LEM/LAB is jettisoned and the SPS is utilized to establish a descent trajectory. Re-entry and recovery takes place in the primary recovery area (see 24).

3.1.2.2 Configuration Summary

The launch vehicle used for this type of mission will be the Saturn V, consisting of three propulsive stages (S-IC, S-II, and S-IVB) and a non-propulsive module, the instrument unit (IU) (see 101-6).

The spacecraft for this mission will use the Apollo Block II CSM, a modified LEM, a LES and an SLA, with added expendables and subsystems to permit operations for up to 45 days (see 101-2) at a mission altitude of 19,350 n. mi.

3.1.2.3 Experiments Summary

Fourteen types of experiments in the fields of astronomy and bioscience are planned for this flight. The primary experiments of Mission 3 are high-resolution astronomical photography and associated calibration photometry (in the X-ray, visual, and infrared portions of the spectrum), experiments with living organisms, and study of the nature of the conjugate auroras.

The telescope, located in the descent stage of the LEM, accounts for about 3700 pounds of the 6000-pound total experiment equipment weight. The total volume required for experiment equipment is 119 cubic feet, and the average power required is 535 watts, with peaks to 1050.

3.1.3 MISSION OBJECTIVES

3.1.3.1 Experiments and Objectives

The experiment objectives for Mission 3 are the gathering of data through:

a. Evaluations of space operational capabilities by deployment of a small maneuverable satellite, a solar balloon, and erection of a radio astronomy antenna.
b. High-resolution astronomical photography and associated calibrated photometry in the X-ray, visual, and infrared portion of the spectrum.
c. Investigation of the nature of the conjugate auroral forms and geometries in the Arctic and Antarctic auroral regions.
d. Experiments in bioscience (study of the human in a space environment and of spaceborne microorganisms).

The following experiments will be conducted during Mission 3 (see 32):

a. 0100 Biomedical (excluding 0101 and 0102).
b. 0200 Behavioral.
c. 0401 Microbial Genetics and Growth.
d. 0402 Microbial Development Biology.
e. 0406 Soft Capture, Enumeration, and Identification of Spaceborne Microorganisms.
f. 0501 Radiation Environment Monitoring.
g. 0701 Emission Line Radiometry.
h. 0702 Intermediate Size Reflecting Telescope.
i. 0703 Manned Coronagraph.
j. 0705B Radio Astronomy.
k. 0801 Conjugate Aurora.
l. 1001 Small Maneuverable Satellite.
m. 1303 Extendable Rod Performance Test.
n. 1304 Solar Sailing.

3.1.3.2 Operations Objectives

The Saturn V launch vehicle will be utilized (see 9-1, 9-10). Its objective is to place the spacecraft payload into an elliptical orbit having a 5300-n. mi. perigee and a 19,350-n. mi. apogee (see 48). The orbit will permit the CSM to transpose and dock, and utilize its SPS to achieve a circular orbit of 19,350 n. mi. at zero degrees inclination over the first of the three subsatellite longitudinal points designated for this mission mode.

The spacecraft objectives are to carry the mission experiments and the crew of three to the specified mission orbit, return the crew and data samples, and provide operational support for the crew and experiments during the 45-day mission (see 1).

The crew will consist of three astronauts (see 9-4), who are expected to participate in the performance of experiments beyond merely checking out and operating equipment. They will help ensure the scientific and technical validity of experiments and provide a partial evaluation of results (see 13).

3.1.3.3 Ground Support Objectives

The mission command responsibility will remain the same as for the basic Apollo mission, with the ability to pass the mission command to the onboard crew commander for all but the critical mission phases (see 1).

The normal complement of equipment at Pad 39A (see 7) will be utilized with the Launch Control Center for this mission. Modifications to launch facilities and checkout operations will be minimized (see 9-13).

All MSFN remote sites supporting this mission will communicate directly with MCC in Houston, with sufficient data routed to MCC in Houston in real time to preclude the necessity of remote site controllers (see 15-11). Tracking and communications are required continuously throughout the Saturn V powered flight (see 15-66).

The mission recovery operations will be similar to those of the basic Apollo system, with the Pacific Recovery Range remaining the primary recovery area (see 14). The potential use of recovery/rescue aircraft will be considered in lieu of continuous assignment of ships for world-wide coverage of abort conditions (see 9-70). When return is initiated from the last longitudinal location, recovery is planned for the nominal Apollo Pacific Recovery Area. Alternate recovery coverage must be developed for early returns from the two preceding longitudinal locations.

The objectives for post-flight operations, involving such activities as data reduction, equipment performance review, crew performance and ground support performance reviews, have not been considered in the AAM program documentation.

3.1.4 MISSION GROUND RULES

3.1.4.1 Earth Launch Phase

Launch Complex 39A, Merritt Island Launch Area (MILA), will be used for the launch of Mission 3 (see 1-31).

The range safety launch azimuth limits will be 72 to 108 degrees based on the earth-fixed heading of the launch vehicle at the beginning of the pitch program (see 29-1). Launch azimuth for this mission is 90 degrees (see 19). Continuous tracking is required from launch to three minutes after insertion into the initial orbits (see 91).

3.1.4.2 Parking Orbit Phase

The parking orbit lifetime will be at least ten orbits, based on the ARDC 1962 atmosphere. The minimum acceptable perigee altitude will be 85 n. mi. The nominal parking orbit for use in initial trajectory studies will be a near-circular orbit of 100 n. mi. altitude (see 29-1).

Injection will occur at the earliest opportunity after spacecraft systems are checked out and the orbit ephemeris is determined (see 29-2). MSFN will be the primary source for trajectory determinations (see 26-5).

A minimum of two station contacts during the first parking orbit and one station contact during all subsequent parking orbits is required (see 14-33). For station contact considerations, a contact time of at least four minutes at an elevation angle of five degrees or higher is required (see 91).

3.1.4.3 Transfer Trajectory Phase

Each of the transfer trajectories will be a Hohmann ellipse (see 19). The astronauts will have control of the S-IVB reaction jets, for attitude control, from earth orbit insertion to separation of the CSM from the S-IVB/LEM (see 24-2). Docking maneuvers are controlled manually. All astronauts will be in space suits for all major velocity maneuvers (see 24-3).

The launch vehicle high-gain antenna pattern will be oriented to an MSFN station, and voice communication will be maintained throughout the period. No ground rules on communications during transposition and docking can be written at this time.

Vehicle illumination will be controlled through changes in yaw. There will be no natural lighting or radiation constraints on transposition and docking (see 29-3) other than the requirement for sun illumination of the LEM docking tunnel and the requirement that the sun vector angle with the CSM X-body be less than 90 degrees.

One four-minute preburn station contact is required within 90 minutes before burn; one four-minute post-burn contact is required within seven minutes after burn (see 91).

3.1.4.4 Mission Orbit Phase

The nominal mission orbit will be a near-circular orbit of 19,350-n. mi. altitude at a 0-degree inclination (see 19).

The spacecraft will maintain observation stations at three sites. The separation of the first and third sites in the direction of drift will be greater than 180 degrees (see 48).

Flight mission is under control of the flight director who will, under normal circumstances, delegate control of the conduct of the scientific program after the spacecraft has achieved its designated flight path (see 14-55). Time between two successive MSFN contacts should not exceed two hours (see 91).

The design of the spacecraft permits extravehicular transfer of the crew between CM and LEM in either direction, in both the docked and undocked condition (see 24-3).

The SPS will be used for de-orbit (see 14-63). The SM RCS system will be utilized for station keeping and shifting (see 49). At least one astronaut in the CM will be wearing a spacesuit at all times during flight. This may be with the faceplate open or closed (see 24-2). No more than two astronauts will be off duty at any one time during the mission (see 76-5).

Ground rules for experiment scheduling cannot be provided at this time. Sample scheduling has been prepared as described in paragraph 3.2.3, which provides the crew task data for experiments, including scheduling, priorties and time allotments for the various experiment functions.

3.1.4.5 Re-entry Phase

The vacuum perigee will be 26 ±14 n. mi. (see 49-8). The re-entry corridor is defined by the negative lift "overshoot" boundary and an "undershoot" boundary, which permits guided re-entry trajectories to targets up to 2500 n. mi. (earth reference) from the re-entry point (see 29-7).

The minimum re-entry range from 400,000 feet altitude to touchdown will be 1500 n. mi. A spacecraft aerodynamic maneuver capability of at least 1000 n. mi. is required throughout the re-entry corridor to provide a weather avoidance capability.

In the event of re-entry guidance or navigation system failure, re-entry monitoring and backup trajectory control techniques will limit the re-entry range to a maximum value of 2500 n. mi. (see 29-7).

No apparent restrictions are involved in return from equatorial orbit. All re-entry trajectories will be posigrade (see 29-8). There are no lighting constraints on earth landings (see 29-8).

The nominal re-entry trajectory will not require cross-range maneuver capability (see 29-8); hence, it will be in the equatorial plane. One 4-minute station contact is required within 90 minutes prior to retro; one 4-minute contact is required not later than 7 minutes after retro (see 91).

3.1.4.6 Earth Recovery Areas

The earth touchdown points will be at the equator and in the primary recovery area (see 24).

3.1.5 MISSION WEIGHTS SUMMARY

The spacecraft weights for Mission 3 are summarized in Table 3-1. The table is broken into three sections: command module, service module, and LEM/LAB. For each module the major categories are detailed and referenced. The command module is an NAA Block II modification (see 45).

The service module is an NAA Block II modification, Configuration D (see 45), except for RCS fuel loading, which is based on Grumman AES studies (see 32), and the SPS

Table 3-1

Mission No. 3 - Synchronous Observation Post
Flight No. 516 - Spacecraft Mission Orbit Injection Weights Summary

Module	Weight (Pounds)		Reference Number and Page
1. Command Module			
Empty Weight	9,034		(45) No page
Useful Weight	1,269		(45) No page
Total	[10,303]		
2. Service Module			
Empty Weight	8,921		(45) No page
Reaction Control Propellant[1]	2,457		
Experiments RCS Propellant		(688)	(31) 1.10.28
Excess RCS Propellant		(1769)	
Cryogenic Expendables	653		(45) No page
SPS Retro Fuel[2]	14,554		Part II, Sect. 6
SPS Fuel[3]	3,295		
Total	[29,880]		
3. LEM/LAB			
Structure	2,549		(32) 1.10.28
SCS	43		(32) 1.10.28
NGS	120		(32) 1.10.28
Crew Provisions	309		(32) 1.10.28
RCS	1,650		(32) 1.10.28
(Propellant)		(1150)	(32) 1.10.28
Controls & Displays	606		(32) 1.10.28
Environmental Control	1,002		(32) 1.10.28
Instrumentation	151		(32) 1.10.28
EPS	4,896		(32) 1.10.28
(Reactants and O_2)		(2545)	(32) 1.10.28
Communications	32		(32) 1.10.28
Experiments	6,062		(32) 1.10.28
Spares	96		(32) 1.10.28
Total	[17,516]		
Total Spacecraft Weight[4]	57,699		
Maximum Spacecraft Weight into Orbit	61,441		Tables 3-7
Margin	3,742		

1. SM RCS tank capacity, 2457 pounds.
2. This figure includes a 10-percent ΔV reserve.
3. Available for subsatellite position change (see Table 3-7) based on an SM fuel load of 41,000 pounds at liftoff.
4. Summation of all square bracketed numbers.

retro fuel load, which is based on computations assuming impulse burns and an additional 10 percent ΔV margin included in the retro fuel budget (Part II, Section 6). LEM/LAB weights are based on GAEC experiment assignments (see 32).

3.1.6 EXPERIMENT LOAD DESCRIPTION

3.1.6.1 General

The experiment loads for Mission 3 are summarized in Table 3-2. The total weight requirement for experiment equipment is about 6000 pounds; more than half of this weight is for a telescope located in the descent stage space of the LEM.

3.1.6.2 Support

Support data for extended experiments begun during Missions 1 and 2 can be obtained for Mission 3 from Mission 2, paragraph 2.1.5.

The coronagraph experiments require simultaneous measurement of the total solar energy profile at the time of coronal observation, and correlation to the data collected at earth stations, and OSO and AOSO satellites. The equipment consists of an Evans coronagraph, which utilizes an external occulting disk to block the bright light of the sun disk, thereby permitting the Lyot telescope behine the occulting disk to detect features in the adjacent area of the sky as much as 100 million times fainter than the sun.

The conjugate auroras experiments are coordinated with observations from ground stations and satellites.

The operations involving the small maneuverable satellite (SMS) provide the capability of performing several experiments, including those concerning occulting bodies, two satellites of accurately known separation, spatial surveys, and radiation assessments. A small satellite that contains propulsion and can be observed, controlled, and located by an astronaut will be provided.

In the extendible rod performance tests, gradient gravity force and solar perturbation data will be obtained during payout and deployment of extendible members. Malfunctioning will be diagnosed after systems retrieval.

The solar sailing experiments entail the deployment of a 50-foot diameter spherical balloon tethered to the spacecraft, control and response, and sensing equipment.

Table 3-2

Experiment Load Summary

32-1 32-1 32-1

Experiment Number	Experiment Category/ Abbreviated Title	Total Wt. (Pounds)	Volume ($Feet^3$)	Volume Reference	Power (Watts)	
					Average	Peak
0100	Biomedical - (Excluding 0101 and 0102)	301	14.5	(32-1-10.6)	100	50
0200	Behavioral	20				
0401	Microbial Genetics and Growth	60.1	7.7	(32-1-10.6)	30	50
0402	Microbial Development Biology	174.7	21.5	(32-1-10.6)	340	
0406	Soft Capture, Enumeration, and Identification of Spaceborne Microorganisms	43	2.0	(32-1-10.7)	110	25
0501	Radiation Environment Monitoring	46.5	2.64	(32-1-10.6)	6.3	
0701	Emission Line Radiometry	220	4.5	(32-1-10.8)	120	
0702	Intermediate Size Reflecting Telescope	3680	18.5	(32-1-10.8)	200	
0703	Manned Coronagraph	262	8.1	(32-1-10.8)	133	
0705B	Radio Astronomy	75	3.3	(32-1-10.8)	150	
0801	Conjugate Aurora	245	6.1	(32-1-10.8 &.9)	180	
1001	Small Maneuverable Satellite	431	10.2	(32-1-10.9)	110	500
1303	Extendible Rod Performance Tests	269	8.0	(32-1-10.9)	30	20
1304	Solar Sail	235	12.0	(32-1-10.10)	10	
	Totals	6062.3	119.04		535	

Relative motions between the balloon and the spacecraft will refine design data relating force to material surface properties.

3.2 OPERATIONS

3.2.1 MISSION PHASE DESCRIPTION

3.2.1.1 General

Mission 3 analysis and description consists of ten major phases. Table 3-3 shows each major phase, its initiating event, the initiating event time, the concluding event, and the time duration of each phase. The phase descriptions delineate the operations conducted and the crew activities performed during each phase.

Note that some parameters currently do not have assigned values. These parameters were included, based on comparable work in the AMPTF design reference mission, to establish the basis for future development of the mission phase descriptions. A blank space bounded by parentheses where values can be placed will identify these parameters. Details of the functions during the experiment activities are outlined in Appendix D. Scheduling of experiments and flight crew operations are shown in paragraph 3.2.3.

The detailed phase descriptions* for Mission 3 are presented below. Appendix C documents the crew activity information that was adapted from AMPTF data.

3.2.1.2 Phase 1: Preflight

The preflight phase begins with the securing of the launch umbilical tower (LUT) to the launch pad and concludes with launch vehicle liftoff at T -0.

The LUT is secured to the pad, all interfaces are checked, all space vehicle-to-ground facilities are verified.

The space vehicle checkout begins with a checkout of the GNS. Included in this subphase are launch vehicle propellant tanking tests, verification of all interface connections, spacecraft fueling operations, and a simulated spacecraft flight. All spacecraft systems are then shut down at T -120 hours.

*This description is based on References 24, 26, and 29 and the summary chart of the AAM Earth Orbit Mission, Table 1 of this report.

Table 3-3

Mission 3 - Synchronous Observation Post

Phase	Initiating Event	Event Time Hr Min Sec	Concluding Event	Duration ΔT Hr Min Sec
1.0 Preflight	T -485 hrs	T -485 hrs	Liftoff (T -0)	485 hrs
2.0 Earth Ascent	Liftoff (T -0)	0	Parking Orbit Insertion - S-IVB Cutoff	677.30 sec (1)
2.1 S-IC Thrusting	Liftoff (T -0)	0	S-IC Staging	162.37 sec (1)
2.1.1 Coast		158.57 sec (1)		3.8 sec (1)
2.1.2 S-IC Staging		162.37 sec (1)		
2.2 S-II Thrusting	S-IC Staging	162.37 sec (1)	S-II Staging	371.82 sec (1)
2.2.1 LES Jettison		193.37 sec		
2.2.2 Coast		534.19 sec		4.80 sec (1)
2.2.3 S-II Staging		538.99 sec (1)		
2.3 S-IVB Thrusting	S-II Staging	538.99 sec (1)	Parking Orbit Insertion - S-IVB Cutoff	138.31 sec (1)
3.0 100-N. Mi. Earth-Parking Orbit	Parking Orbit Insertion - S-IVB Cutoff	677.30 sec (1)	TOI - S-IVB Ignition	102 min (3)
4.0 First Transfer Orbit	TOI - S-IVB Ignition	1 h, 53 m, 17 s	Second TOI - S-IVB Ignition	320 m, 24 s
4.1 S-IVB Thrusting	TOI - S-IVB Ignition	1 h, 53 m, 17 s	S-IVB Cutoff	264 sec
4.2 Transfer Orbit Coast	S-IVB Cutoff	1 h, 57 m, 41 s	Second TOI -	316 min

Table 3-3

Mission 3 - Synchronous Observation Post (Cont.)

Phase	Initiating Event	Event Time Hr Min Sec	Concluding Event	Duration ΔT Hr Min Sec
5.0 Second Transfer Orbit	Second TOI - S-IVB Ignition	7 h, 13 m, 41 s	EOI - SPS Cutoff	13.7 hr
5.1 S-IVB Thrusting	Second TOI - S-IVB Ignition	7 h, 13 m, 41 s	S-IVB Cutoff	61.9 sec
5.2 Second Transfer Orbit Initial Coast	S-IVB Cutoff	7 h, 14 m, 43 s	CSM/LEM Separation	13.7 hr
5.3 Transposition and Docking	CSM/LEM Separation		Hard Dock	40 min(2)
5.4 S-IVB Staging	Hard Dock	7 hr, 45 min	Staging of S-IVB/ IU/SLA	5 min
5.5 Second Transfer Orbit Coast	Staging of S-IVB/ IU/SLA	7 hr, 50 min	EOI - SPS Ignition	
5.5.1 Midcourse Correction				
5.6 EOI - SPS Thrusting	EOI - SPS Ignition	20 hr, 57 min	EOI - SPS Cutoff	5 min, 4 sec
6.0 Synchronous Orbit Coast		21 hr, 2 min		~44 days
6.1 First Observation Post				
6.1.1 Insertion - 126 degrees west	Insertion at 1st Post	21 hr, 2 min	End Observation	~10 days
6.1.2 Transition(1)	Begin Transition to 2nd Post	11.0 days	Complete Transition	3.8 days
6.2 Second Observation Post				
6.2.1 Insertion - 70 degrees west	Insertion at 2nd Post	14.8 days	End Observation	10 days
6.2.2 Transition(2)	Begin Transition to 3rd Post	24.8 days	Complete Transition	11.5 days

Table 3-3

Mission 3 - Synchronous Observation Post (Cont.)

Phase	Initiating Event	Event Time Hr Min Sec	Concluding Event	Duration ΔT Hr Min Sec
6.3 Third Observation Post				
6.3.1 Insertion - 90 degrees east	Insertion at 3rd Post	36.3 days	End Observation, Begin De-orbit Preparations	8.7 days
7.0 De-orbit	Beginning of De-orbit Preparations	45 days	Re-entry	6 hr, 49.5 min
7.1 LEM Staging	Beginning of De-orbit Preparations		SPS - Retro-ignition	1 hr, 30 min(4)
7.2 SPS - Retrothrusting	SPS - Retroignition		SPS - Retro Cutoff	3.5 min
7.3 De-orbit Coast	SPS - Retro Cutoff	45d, 1hr, 33.5m	SM Staging Initiation	4 hr, 21 min
7.4 SM Staging	SM Staging Initiation	45d, 5hr, 54.5m	Re-entry	55 min
8.0 Re-entry	400,000-Feet Re-entry	45d, 6hr, 49.5m	Drogue Chute Deployment	10.5 min(2)
9.0 Parachute Descent	Drogue Chute Deployment	45d, 7hr	Touchdown	7 min(2)
10.0 Recovery and Retrieval	Touchdown	45d, 7hr, 7m	Retrieval	

(1) Reference 49, Table 11.
(2) Reference 24, page 4.13-3.
(3) Injection at 3rd equatorial crossing.
(4) AMPTF adapted.

The pre-countdown begins 485 hours before the scheduled launch time and continues until terminal countdown, 10 hours prior to launch. During this period the LEM and CSM receive cryogenic servicing, the GNS of both the CSM and LEM are checked and activated, and the launch vehicle is loaded with propellant. Other operations include the installation of cabin, crew, and water supplies, and the final installments of LEM scientific equipment.

The terminal countdown begins ten hours prior to launch. The CSM is cleaned, sterilized, and closed out. The launch vehicle obtains its cryogenic load, while tracking and communications networks are tested. At T -6 seconds, the S-IC engine ignition sequence is started and at T -0, the vehicle lifts off, ending the preflight phase.

3.2.1.3 Phase 2: Earth Ascent

The earth ascent phase begins with Saturn V liftoff at T -0 seconds and concludes with the S-IVB cutoff after earth parking orbit insertion.

After a six-second thrust buildup period from S-IC ignition, the vehicle is released and lifts off. The vehicle rises vertically (pitch = 90 degrees) for 12 seconds, and then begins a programmed pitch maneuver that smoothly reduces the pitch angle to 23 degrees at the burnout point. At 30,000 feet, the dynamic pressure reaches a peak of 690 psf. At S-IC staging, T +162.4 seconds, the vehicle velocity and approximate altitude are 8992 fps and 170,000 feet. The S-IC booster is staged prior to S-II ignition.

During ascent the commander monitors acceleration, attitude, attitude rates, abort lights, and pitch and roll rates. The navigator is responsible for monitoring the S-IC ignition sequence, and the caution and warning indicators. The navigator also monitors the environmental control system panel. For the duration of the ascent phase, the engineer maintains radio communications with the control center and monitors the critical temperature flow and events displays. In addition, the commander reports the S-IC engine cutoff and staging sequence using the two-way voice communication link.

After S-IC cutoff, the vehicle coasts 3.8 seconds before S-II ignition at T +162.4 seconds. The pitch angle is changed from 23 to 31 degrees prior to LES jettison, which occurs 31 seconds after S-II ignition. With the completion of the LES jettison, the vehicle assumes a nearly linear pitch variation, which reduces the pitch angle to approximately 0 degrees at S-II burnout. The commander confirms the LES jettison 31 seconds after verifying S-II ignition. The S-II engines shut down approximately nine

minutes after launch, upon a signal from the propellant level sensors. The commander monitors this cutoff and staging sequence.

S-II cutoff T +534.2 seconds, with a vehicle velocity and altitude of 22,731 fps and 606,632 feet, respectively. At this point, the S-II is staged and the CSM/LEM-S-IVB configuration is allowed to coast for 4.8 seconds before S-IVB ignition occurs.

The S-IVB is ignited at T +539 seconds. The vehicle performs a programmed pitch maneuver that terminates with the vehicle's insertion into a 100-n. mi. earth-parking orbit and cutoff of the S-IVB. S-IVB cutoff occurs at T +677.3 seconds; vehicle velocity at S-IVB cutoff is 25,582 fps.

The crew continues to monitor vehicle performance during the first S-IVB burn. The guidance system commands the S-IVB cutoff after the achievement of a 100-n. mi. earth orbit.

3.2.1.4 Phase 3: Earth Parking Orbit

The 100-n. mi. earth-parking orbit phase (see Appendix C) begins with the cutoff of the S-IVB and ends with the ignition of the SPS in order to permit transfer into synchronous orbit. The parking orbit is circular with an inclination of 28.5 degrees.

After S-IVB engine cutoff, the commander and navigator confirm the achievement of the 100-n. mi. parking orbit. All three crewmen then adjust their couch restraints and set the controls and displays for parking orbit operations. Using the inflight test system, the navigator and engineer perform a checkout of spacecraft systems, including guidance and control, environmental control, reaction control, and electrical power. The engineer also continues to maintain communications with MCC.

The commander then compares with MSFN the onboard calculations of the proposed Hohmann injection maneuver to the present trajectory parameters. In preparation for the transfer orbit insertion, the navigator performs both coarse and fine IMU alignments.

After stabilizing the spacecraft attitude for these alignments, the commander maneuvers the vehicle to the desired attitude for the second S-IVB insertion burn into the Hohmann transfer orbit. The navigator programs the guidance computer for transfer insertion, secures the controls and displays, and monitors the S-IVB ullage sequence.

The engineer prepares the SPS system for a quick-abort contingency ability. At mission time 1 hour 53 minutes 17 seconds, longitude -30 degrees, and latitude 5 degrees, the S-IVB automatically ignites.

3.2.1.5 Phase 4: First Transfer Orbit

The first transfer orbit coast phase (see Appendix C) begins with S-IVB thrusting to transfer the spacecraft from its 100-n. mi. parking orbit to its synchronous altitude transfer ellipse. Termination of Phase 4 occurs when the S-IVB is again thrusted to initiate a partial circularization of the transfer ellipse.

At 102 minutes after parking orbit insertion, the CSM GNS automatically fires the S-IVB to place the spacecraft in a Hohmann transfer orbit with an apogee of 19,350 n. mi. Two degrees of plane change are included in the maneuver so that the transfer trajectory inclination is 26.5 degrees. Thrusting is initiated at a geocentric longitude and latitude of -30 degrees and 50 degrees, respectively. After 264 seconds of thrusting the S-IVB is shut down.

All three crewmen monitor the S-IVB ignition and second burn into the Hohmann transfer orbit. The commander continues to monitor the guidance and navigation attitude displayed on the FDAI for the duration of this 264-second thrust impulse. The crew confirms S-IVB cutoff.

After the navigator verifies orbit achievement, the commander sets the guidance and control system for transfer orbit operations. Once he has computed and confirmed the orbit parameters, the commander makes the decision for mission continuation in the normal mode.

The navigator and engineer check out the electrical power and SM reaction control systems. During the five-hour transfer orbit coast to the second transfer orbit, the navigator again performs coarse and fine IMU alignments and takes several navigational sightings. The guidance computer is used to calculate the required transfer orbit parameters by using data obtained from the IMU alignment. The commander and navigator compare these onboard calculations with MSFN tracking values and establish the intermediate orbit insertion point. After completing a CSM system status check, the crew secures couch restraints in preparation for the second transfer orbit insertion maneuver.

The spacecraft coasts in the transfer orbit. At apogee the CSM GNS automatically ignites the S-IVB for a transfer orbit adjustment. With this event, the first transfer orbit coast phase is terminated.

3.2.1.6 Phase 5: Second Transfer Orbit Coast

Phase 5 begins at mission time T +7 hours and 13 minutes, 83 degrees longitude, and 0 degrees latitude with S-IVB transfer orbit adjustment thrusting (see Appendix C). The phase ends with the insertion of the spacecraft into the synchronous earth orbit by SPS thrusting.

The S-IVB is powered for 61.9 seconds to adjust the spacecraft transfer orbit into an elliptic trajectory with a perigee of 5294 n. mi. and an apogee of 19,350 n. mi. The S-IVB includes a 17.4-degree plane change in the maneuver so that the second transfer ellipse has an inclination of 9.1 degrees. All three crewmen monitor the second transfer orbit insertion at T +7 hours and 14 minutes.

With the termination of S-IVB thrusting, at a mission time of 7 hours 14 minutes 43 seconds, the commander and navigator confirm the transfer orbit achievement, and the crew adjusts restraints. The engineer continues to record the PCM telemetry and maintain communications with MSFN. Once the navigator and engineer have completed a CSM systems status check, the commander orients the spacecraft for an IMS fine alignment and landmark navigational sightings.

After the navigator performs these navigational sightings, the commander verifies the onboard trajectory information using ground-computed values. The crew then prepares for the transposition and docking maneuver.

The spacecraft is oriented for transposition and docking. After the S-IVB SCS is placed in the attitude hold mode, the commander initiates the transposition maneuver by actuating separation of the S-IVB from the CSM.

When the S-IVB-LEM/CSM separation is assured, the SLA is deployed to aid in the stabilization of the S-IVB-LEM and to expose the LEM. The CMS then translates 35 feet forward, performs a 180-degree pitch maneuver, and translates back to the S-IVB-LEM. The CSM crew performs docking maneuvers to align the CM-LEM interface and to engage the LEM drogue (X-axis docking tunnel). A maximum of 15 minutes is require from separation to the accomplishment of a soft dock, which occurs at a mission time of 7 hours 20 minutes. After soft docking has been achieved, the gator secures the docking latches and electrical connections between the two modules

navigator secures the docking latches and electrical connections between the two modules and pressurizes the LEM. A maximum of 25 minutes is required from soft dock to the securing of the CM hatches after performing the hard dock.

Upon completion of hard docking, the commander initiates the S-IVB separation from the CSM/LEM configuration, a sequence which is monitored visually by the engineer. The commander then translates the CSM/LEM from the jettisoned S-IVB stage. The engineer reports the maneuver completion to the Mission Control Center.

A maximum of 5 minutes is required from hard docking to the completion of the S-IVB/CSM-LEM separation. S-IVB staging occurs at a mission time of T +7 hours, 50 minutes, 75 degrees longitude, and 5 degrees latitude.

In preparation for the SPS burn into a synchronous orbit, the commander maneuvers the CSM/LEM for an IMU fine alignment. The navigator performs the IMU alignment and calculates the synchronous orbit trajectory parameters. The commander compares the navigator's calculations with position data received from MSFN and decides whether a midcourse correction is necessary.

Upon completion of any midcourse correction the commander makes a decision for SPS insertion burn. The navigator conducts a final guidance and control system check. The engineer checks out the service propulsion and electrical power systems and the commander verifies the spacecraft attitude. The three crewmen then adjust restraints for the SPS insertion burn into the synchronous earth orbit.

At a mission time of T +20 hours, 57 minutes, the GNS automatically commands SPS ignition. SPS thrusting will be used to place the spacecraft in a circular orbit and to execute a 9.1-degree plane change. SM RCS ullage ignition occurs 5 seconds prior to SPS ignition. This phase is terminated with the insertion of the spacecraft into synchronous earth orbit.

The crew monitors the SPS ignition and burn into a 19,350-n. mi. earth orbit. The commander monitors the guidance and navigation attitude for the duration of this 304-second thrust impulse. The crew confirms SPS cutoff.

3.2.1.7 Phase 6: Synchronous Orbit Coast

The spacecraft is inserted into synchronous earth orbit (see Appendix C) at a mission time of T +21 hours, 2 minutes, and a geocentric location of 19,350 n. mi. altitude, 126 degrees west longitude, and 0 degrees latitude. This phase ends with the beginning of de-orbit preparations.

The navigator confirms synchronous orbit injection using the spacecraft displays, the guidance computer, and data from MSFN. He also secures the controls and displays before assisting the commander in a CSM system status check. While the engineer continues to record PCM telemetry and maintain ground communications, the navigator proceeds with the IMU fine alignment and the landmark navigation sightings. During these operations the commander maneuvers the vehicle attitude for landmark acquisition. The navigator next compares his orbit information with ground computed parameters to ensure accuracy of the onboard calculations.

The engineer prepares to transfer to the LEM for orbit operations. The commander checks the LEM pressurization from the CM. Then, the engineer equalizes the CM and interlock pressure, removes and stows the CM pressure and thermal hatches, and disconnects and stows the probe and drogue. After activating and checking out his portable life support system, the engineer transfers from the CM to the LEM by proceeding through the tunnel. Upon completion of an inspection of the LEM cabin, the engineer closes the hatch and initiates communications with the MSFN. After a brief checkout period, experiments that require crew activation or participation are set up.

During each day of the orbit phase, the crew's primary activities include scientific experiments, system status checks, and navigational sightings. Personal functions are performed in the time remaining.

On the eleventh mission day, at a mission time of 11 days, the SM SPS is thrusted for 12 seconds to change the post of the synchronous satellite. After a coast period of 3.8 days, the SM SPS is retro-thrusted to replace the CSM/LEM configuration into a synchronous earth orbit at 70 degrees west longitude and 0 degrees latitude.

With retro-thrusting terminated, the crew returns to its normal work/rest cycle. Experimentation is continued until the 24.8 mission day. At this time, the crew prepares for a second post changing maneuver. At mission time, T +24.8 days, the SM SPS is

automatically ignited by the CM-GNS. Thrusting continues for 12 seconds. The CSM/LEM coasts for 11.5 days before SPS retro-thrusting replaces the spacecraft in synchronous earth orbit at 90 degrees east longitude and 0 degrees latitude.

Crew rest/work cycles continue until the last mission day. At the end of the 45-day orbit phase of the mission, the engineer deactivates the LEM systems and transfers to the CSM. At this time, the crew transfers to the LEM all CM and experimental equipment that is to be jettisoned. Upon completion of these activities, each crew member enters his couch, fastens his restraints, and begins preparation for the LEM jettison. The synchronous earth orbit coast phase is terminated with these preparations.

3.2.1.8 De-orbit

The de-orbit phase begins when the crew starts de-orbit preparations and terminates with the CM entry into the earth atmosphere at 400,000 feet.

The LEM is staged at a mission time of T +45 days at a geocentric location of 19,350 n. mi. altitude, 90 degrees east longitude, and 0 degrees latitude.

After orienting the spacecraft for LEM jettison, the commander initiates LEM/CSM separation by igniting shaped charges to sever the LEM/CSM interface and then translates the CM from the LEM by using the SM reaction control system. The commander maneuvers the spacecraft to the proper attitude while the navigator performs a navigational sighting. After completing the IMU fine alignment and a landmark navigational sighting, the navigator computes the onboard trajectory. MSFN supplies orbital parameters for comparison with the navigator's trajectory calculations. The crew then checks all CSM systems in preparation for the third SPS burn into the de-orbit coast phase.

Following verification of the navigator's de-orbit parameters by MSFN, the three crewmen don their pressure suits and adjust their seat restraints.

The CSM-GNS is programmed and the CSM is oriented before the crew initiates the countdown for SPS retroignition. At the termination of the countdown, the CSM-GNS automatically commands SM RCS ullage ignition and SPS ignition at the proper times.

An SPS retrofiring of 210 seconds inserts the CSM into a descending transfer orbit having a vacuum perigee of 26 ±14 n. mi. The SPS thrust initiation begins at a mission time of 45 days 1 hour 30 minutes, and at 90 degrees longitude and 0 degrees latitude.

The commander checks the insertion parameters, the CSM systems status, and communicates his findings to MSFN. Next, he prepares for SM jettison by orienting the CSM to the proper attitude and activating the CM-SM separation sequence. During this period, the engineer switches all systems from SM to CM and retracts the antenna and all optics. After a final IMU alignment, the CSM is properly oriented and the SM is staged. SM staging occurs 55 minutes prior to re-entry. At the time of SM staging, the spacecraft is ( ) miles above the surface of the earth, at ( ) longitude and 0 degrees latitude.

All three crewmen monitor the CM-SM separation. After SM separation, the navigator and engineer perform post-separation status check of the CM-RCS. The commander then maneuvers the CM to the MSFN confirmed entry attitude. Following a check by the commander of the entry monitor flight indicator, the crew members adjust restraints, check suit umbilicals, and close face plates.

Following SM jettison, a final pre-entry check is conducted. The checkout includes an RCS and GNS entry check, an entry monitor systems check, a crew equipment systems check, and a CM orientation for re-entry. The de-orbit phase ends when the CM reenters the earth's atmosphere at a nominal altitude of 400,000 feet. Re-entry occurs at a geocentric longitude and latitude of 162 degrees and 0 degrees, respectively.

3.2.1.9 Phase 8: Re-entry

The re-entry phase begins when the CM enters the earth's atmosphere, and ends at the time of drogue chute deployment, which occurs at an altitude of 24,000 feet.

The re-entry phase begins at a mission time of T +45 days, 6 hours, and 49.5 minutes, at a nominal entry altitude of 400,000 feet above the surface of the earth, and at a geocentric longitude and latitude of 162 degrees and 0 degrees, respectively. For the preselected landing site target of -158 degrees and 0 degrees, the range to go at the time of entry is 2500 n. mi. The entry velocity, flight path angle, inclination, and azimuth are 33,840 fps, -6.4 degrees, 0 degrees, and 90 degrees, respectively.

The initial part of the re-entry phase is a full-lift flight from 400,000 feet to a point where the rate of descent is reduced to 700 fps. At this point, trajectory range control is initiated. During trajectory range control, the steering commands are based on a reference trajectory with the guided trajectory predicted range from the control initiation point to touchdown matching the range requirements to reach the target. The

final part of the entry phase consists of an equilibrium glide to an altitude of 24,000 feet, that is, the altitude at which the drogue chute deployment occurs.

The CM earth landing system is armed during the equilibrium glide when the CM passes through an altitude of 100,000 feet. The forward heat shield is jettisoned at an altitude of 25,000 feet, 3-seconds prior to drogue chute deployment, which marks the beginning of the parachute descent phase.

During re-entry the commander continues to monitor the CM entry attitude and the glide maneuver. The navigator starts the elapse event time clock upon a 0.05-g indication. The engineer maintains communications with the MSFN recovery forces throughout this re-entry phase. At the 100,000-foot altitude, the commander arms the earth landing system and monitors the descent to 25,000 feet.

3.2.1.10 Phase 9: Parachute Descent

The parachute descent phase begins at an altitude of 24,000 feet, at the time of drogue chute deployment, and ends approximately 7 minutes later with CM earth landing. This phase begins at a mission time of T +45 days and 7 hours when the CM is at an altitude of 24,000 feet and approximately over the earth landing site location. The CM relative velocity and flight path angle at this point is ( ) fps and ( ) degrees, respectively.

The navigator monitors the forward heat shield deployment at 25,000 feet and the drogue chute deployment 1.5 seconds later. If parachute deployment does not occur automatically, the parachute is deployed manually. Fifty-eight seconds after the beginning of the phase, the drogue chutes are jettisoned and the main chutes are deployed at an altitude of about 11,000 feet.

After the navigator activates the VHF system, all three crewmen shut off all power switches not required for landing. The remaining propellant in the CM-RCS is then dumped by the navigator. The commander continuously monitors the altitude and descent velocity during this phase, communicating the status of the descent to the MSFN.

Earth landing occurs at 420 seconds after the deployment of the drogue chutes. At CM landing, the relative velocity and flight path angle are 25 fps and 90 degrees, respectively. Earth landing occurs at a longitude of -158 degrees and a latitude of 0 degrees. Recovery communications (VHF) are activated and a final systems check is performed during parachute descent, starting 4 minutes and 49 seconds prior to touchdown.

3.2.1.11 Phase 10: Recovery and Retrieval

The recovery and retrieval phase begins with CM touchdown and continues until both the CM and crew are onboard a recovery ship. The recovery and retrieval phase begins at a mission time of T +45 days, 7 hours, and 07 minutes, the time of CM touchdown.

After touchdown, each crew member releases his seat restraints and assesses his physical condition and the structural condition of the CM. While the commander activates the post-landing recovery aids, the engineer checks out the EPS and ECS for post-landing operations. After obtaining survival equipment from the storage area, the crew egresses to await recovery.

3.2.2 CONFIGURATION

3.2.2.1 General

The command and service modules for Mission 3 are NAA Block II, Modification D, modules (see 45). The LEM/LAB configuration and the location of experiment equipment are shown in Figure 3-1. Conceptual sketches are available for several of the equipment used in the mission and are shown in Figure 3-2.

Equipment for experiments in space structures technology, the conjugate auroras, and the maneuverable satellite are mounted for external access in the LEM descent stage. Gimballed mounts are provided for the extendible access in the LEM descent stage and for the extendible rod antenna of the radio astronomy experiment. The subsatellite is checked out, launched, and recovered by using its launch package, and refueled and refilmed during EVA's. Supporting electronics and spares for experiments are carried in the aft/equipment bay of the ascent stage.

Structural modifications to the LEM ascent stage are limited to the incorporation of an airlock below the floor panel in the forward cabin to accommodate the coronagraph. The telescope air lock is designed to mount in the existing ascent stage engine hatch. The descent stage structure must be reworked to accommodate the coronagraph. Supports for all other items can be added without modification.

In addition to the standard CSM/LEM interface, this flight requires additional wiring to accommodate the pointing requirements of the experiments. Interface wiring includes star tracker inputs to maintain attitude hold, and command data for the laboratory flight controller for manual control of spacecraft slewing operations.

The astronomy experiments are all aligned parallel to the longitudinal axis of the spacecraft and are controlled from an astronomer console on the right side of the forward cabin. Access to the telescope and coronagraph is provided from inside the laboratory to permit changing the camera, magazines, and spectrographs. A common television viewfinder and two boresight star trackers provide inputs for orientation and control. Spacecraft pointing, controlled from the astronomer console, has an accuracy for these experiments of $\pm 1/2$ degree. The telescope, coronagraph, and conjugate aurora experiments are gimballed and incorporate fine-error sensors to meet individual requirements.

Attitude is held within a deadband of ±5 degrees, except during the preceding experiment periods that have specific pointing and stability requirements. Experiment pointing and stability requirements exceeding CSM capability (0701, 2, 3, 5B, 0801) are accommodated by installing the experiment telescopes on gimballed mounts and providing fine guidance and control systems within the telescopes. Spacecraft pointing and stability requirements are accommodated by the CSM up to the propellant capacity. A LEM RCS system is added to the laboratory to supplement the propellant capacity of the CSM.

Adequate storage and work space is available for all experiments. A control console on the left side of the forward cabin is used for the space structures, conjugate aurora, and subsatellite experiments. All food for the mission is stored internally. A seven day supply of lithium hydroxide canisters is stored internally and replaced from external storage during scheduled EVA.

This flight utilizes an Apollo CSM fuel cell electrical generation system, a LEM ECS to provide suit circuit operation in the laboratory during EVA's, and selected components to provide data transmission, systems status, and other appropriate displays.

The large number of rotational maneuvers required by experiments 0801, 0701, 0702, and 0703, in addition to gravity gradient disturbance torques, create large RCS propellant requirements. Four sets of LEM RCS tanks are carried on the Model D version.

3.2.2.2 Configuration History

The configuration history for the synchronous observation post mission is shown in Figure 3-3, which divides the mission profile into four major operation areas: earth ascent, earth parking and transfer orbits, mission orbit posts, and re-entry and recovery. The launch vehicle and spacecraft configurations are depicted as appropriate for each of these major areas. Mission times from the trajectory analysis (see 3.2.5) and mission phasing charts (Table 3-3) are given, relating time to the various configuration changes.

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KEY

No.	Experiment	Name
1		FCA Radiator
2	0705B	Radio Astronomy 1-5 mc Range Using Tunable V-Antenna
3		Boresight Star Tracker
4	0406	Space-Borne Micro-organism Cartridges
5	0703	Manned Coronagraph
6	0702	Intermed-Size Reflecting Telescope Equipt Radiator
7		H_2
8		FCA
9		O_2
10		
11	1001	Small Maneuverable Satellite
12	1303	Extendable Rod Perf. Test
14	1304	50 Foot Balloon W/Coils, Power Supply & Canister
15	0801	Telescope Assy
16	0701	Emission Line Radiometry

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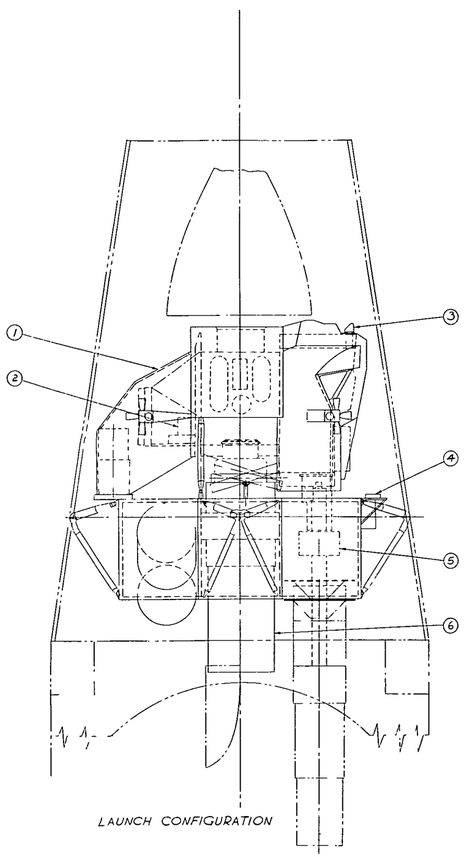

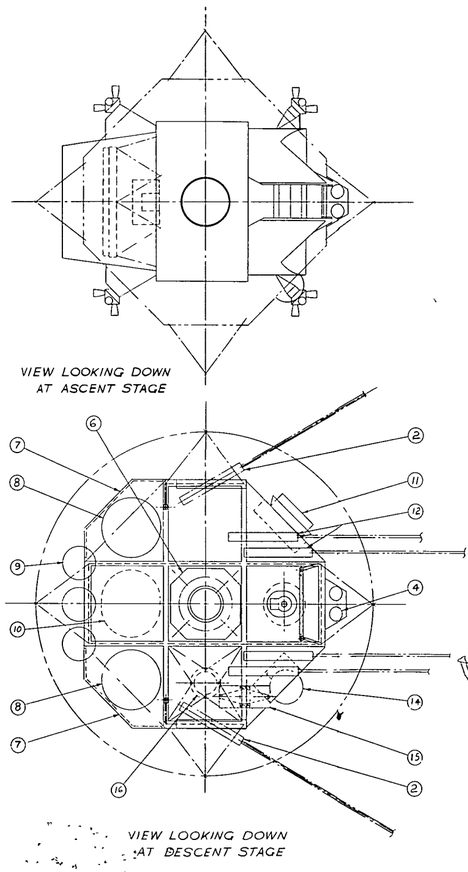

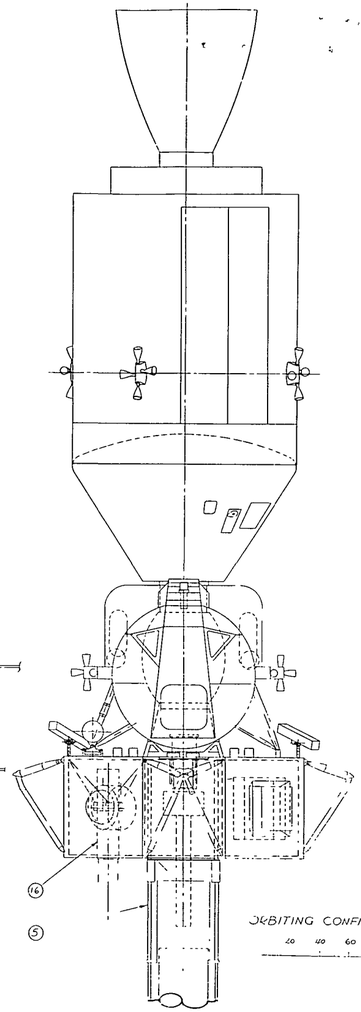

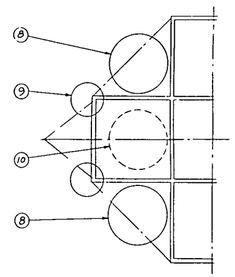

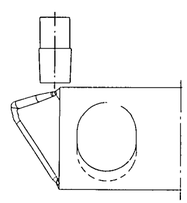

Figure 3-1. Mission 3 - Spacecraft Configuration and Experiment Location (Sheet 1 of 2) (32 pages 1-10, 30)

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3-27-2

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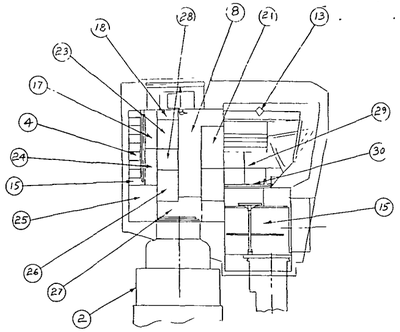
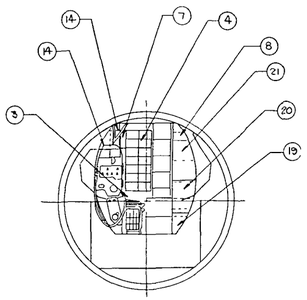
KEY

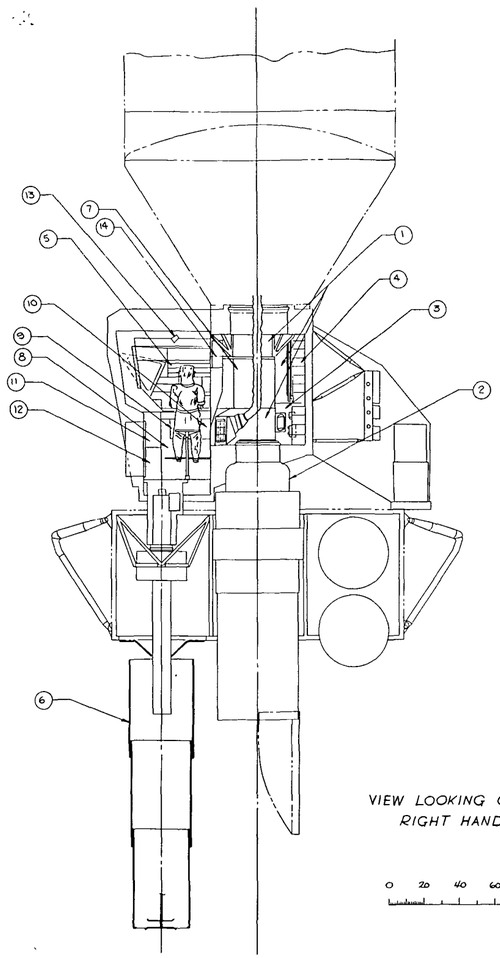
No.	Experiment	Name
1		Food
2	0702	Telescope
3	0402	-20°C Refrigerator
4		LiOH
5		Astronomers Display Console
6	0703	Coronagraph
7		ECS
8		Biomed
9	0100, 0200	Biomed Incubator
10		Biomed Refrigerator
11	0401,0402	Incubator
12	0501	Radiation Environment Monitoring
13		Floodlights (2)
14	1304	Binoculars & Range Finder
15		Logs, Film & Tape Storage, Recorders
16		Seat & Restraint (2)
17	0406	+5°C Refrigerator
18	0401	Lyophilization
19	0401	Chemical Equipment Storage
20	0402	Scintillation Counter
21	0402	Cultrue Flasks
22		ECS & Communications Control Console
23	0401,0402 0406	Microscope
24	0401,0402	-15°C Refrigerator
25	0406	Incubator
26	0402	Particule Counter
27	0402	Support Cabinet & 35mm Camera
28	0402	Centrifuge
29	0801,1001	Display
30	1303,1304 0801,& 1001	Control Console



VIEW LOOKING OUTBOARD
LEFT HAND SIDE

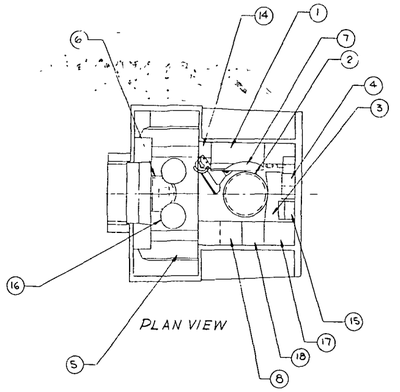


VIEW LOOKING AFT

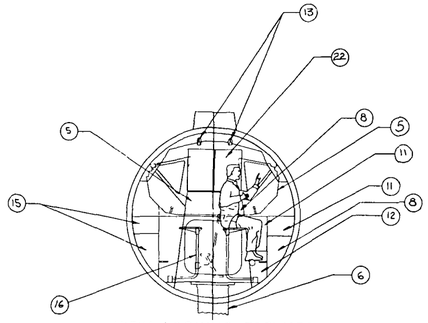


VIEW LOOKING OUTBOARD
RIGHT HAND SIDE

0 20 40 60 80 100



PLAN VIEW



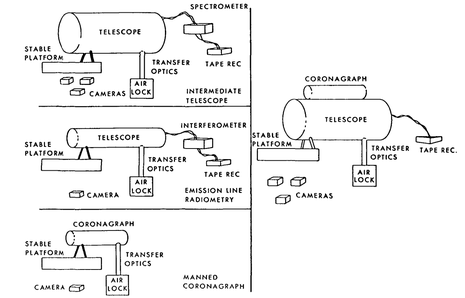
VIEW LOOKING FORWARD

Figure 3-1. Mission 3 - Spacecraft Configuration and Experiment Location
(Sheet 2 of 2)
(32 pages 1-10.31)

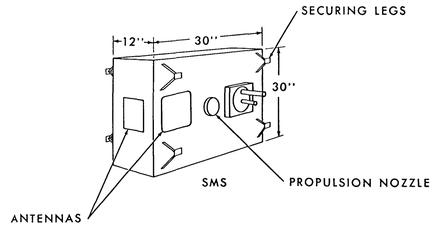
3-28-2

3-28-3

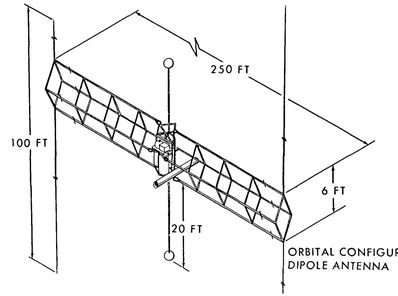
ASTRONOMY EQUIPMENT COMBINATION-0701 0702 0703



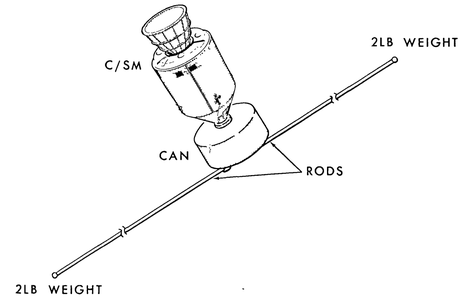
SMALL MANEUVERABLE SATELLITE (SMS) - 1001



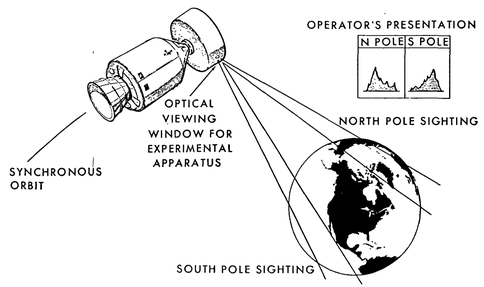
RADIO ASTRONOMY (1-5 MC) - 0705A



EXTENDABLE ROD TESTS - 1303



CONJUGATE AURORA AND AIRGLOW - 0801



SOLAR SAILING PASSIVE COMSAT - 1304

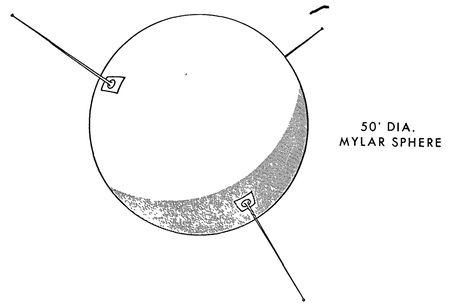


Figure 3-2. Experiment Equipment Conceptual Sketches (See 11, 13)

3-29-1

3-29-2

MISSION 3 SYNCHRONOUS OBSERVATION POST

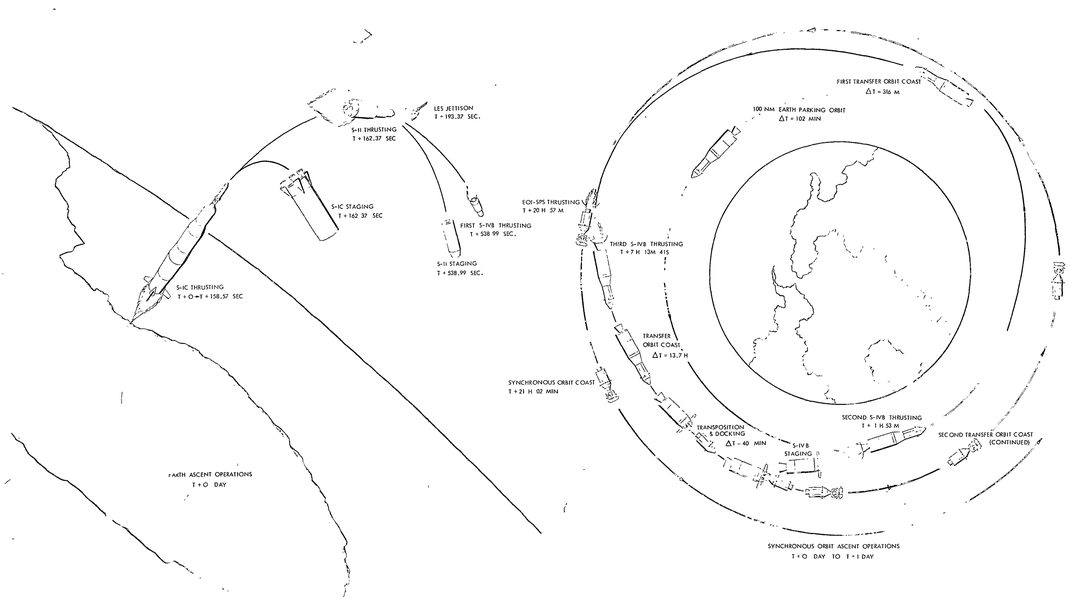
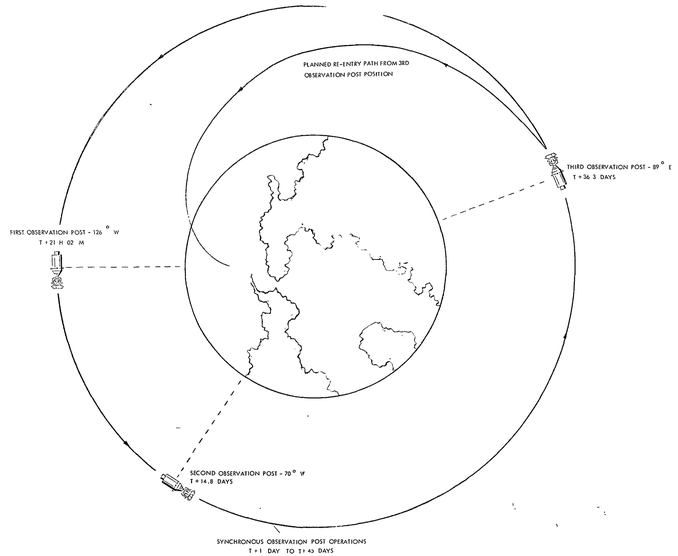
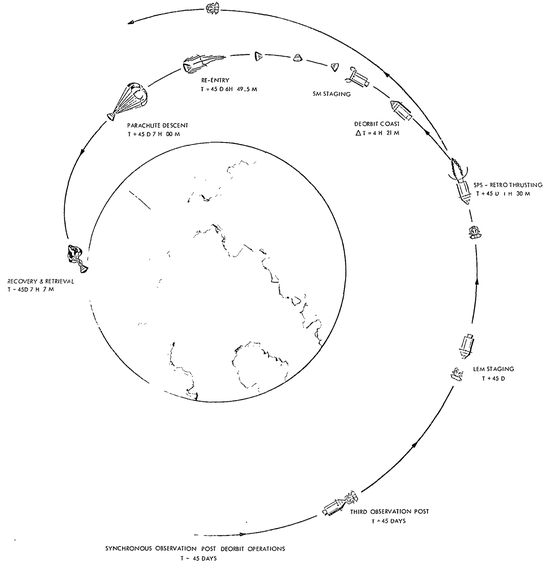


Figure 3-3. Mission 3 - Configuration History





3·30-2

3·30-3

3-30-4

3-305

3.2.2.3 Expendable Weights

3.2.2.3.1 Life Support System

The expendables weights for Mission 3 are summarized in Figure 3-4.

The following expendables are ascribed to Mission 3 (see 126 and 127):

a. Food - 228 pounds are stored on the LEM (see 32-1-10.26). This amount is sufficient to sustain the crew for the mission at a consumption rate of 1.4 pounds per man per day.

b. Oxygen - The CSM has two Block II oxygen tanks filled to 652.8 pounds (see 80-167). The LEM has two Apollo X-tanks loaded to 210 pounds.

c. Hydrogen - The CSM has two hydrogen tanks filled to 58.24 pounds. The LEM has two Apollo hydrogen tanks (see 33-1-13.15) loaded to 226 pounds.

d. Repressurization - For extravehicular activity, cabin atmosphere must be evacuated overboard. It was assumed that the CM and LEM have no airlocks, and that the LEM atmosphere is also dumped. Repressurizations are required (see 32-1-10.16).

e. Leakage - The oxygen leakage rate for the CM is 5.66 pounds per day and for the LEM, 4.8 pounds per day. This results in a total leakage of 276 pounds for the LEM, and 255 pounds for the CSM, based on a cabin pressure of 5 psia.

f. Respiration - The crew oxygen requirements are 1.67 pounds per man per day for a total of 225 pounds, supplied by the LEM.

g. Fuel Cells - The CSM and the LEM each have three fuel cell assemblies (see 80 and 32-1-12.16). The hydrogen and oxygen supply for the cells is the original supply less the amounts required for respiration, repressurization, and leakage replacement.

h. Water - The fuel cell production of water varies, with power requirements, from 28.8 to 82.8 pounds per day for the three cell assemblies combined. Hydrogen and oxygen delivered to the CSM fuel cells combine to form water; excess water is vented overboard.

i. Water of Oxidation - Water is needed to reconstitute the powdered food. The requirement is 0.035 pounds per man per hour for a total of 113.4 pounds.

j. Metabolic Consumption - Drinking water required is 0.0275 pounds per man per hour for a total of 891 pounds.

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CONFIDENTIAL

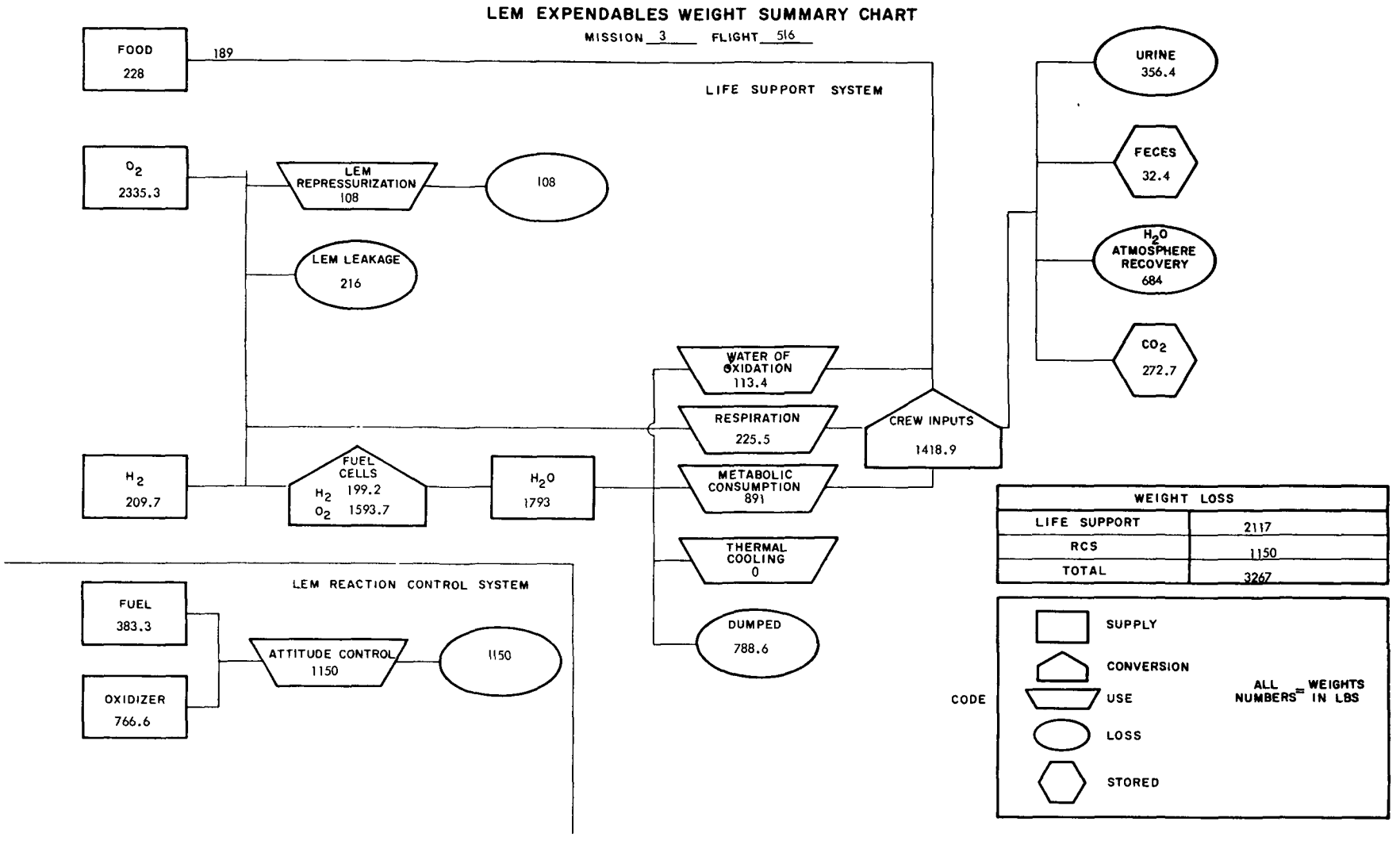


Figure 3-4. Mission 3 - Expendables Weight Summary Chart (Sheet 1 of 2)

CSM EXPENDABLES WEIGHT SUMMARY CHART

MISSION 3 FLIGHT 516

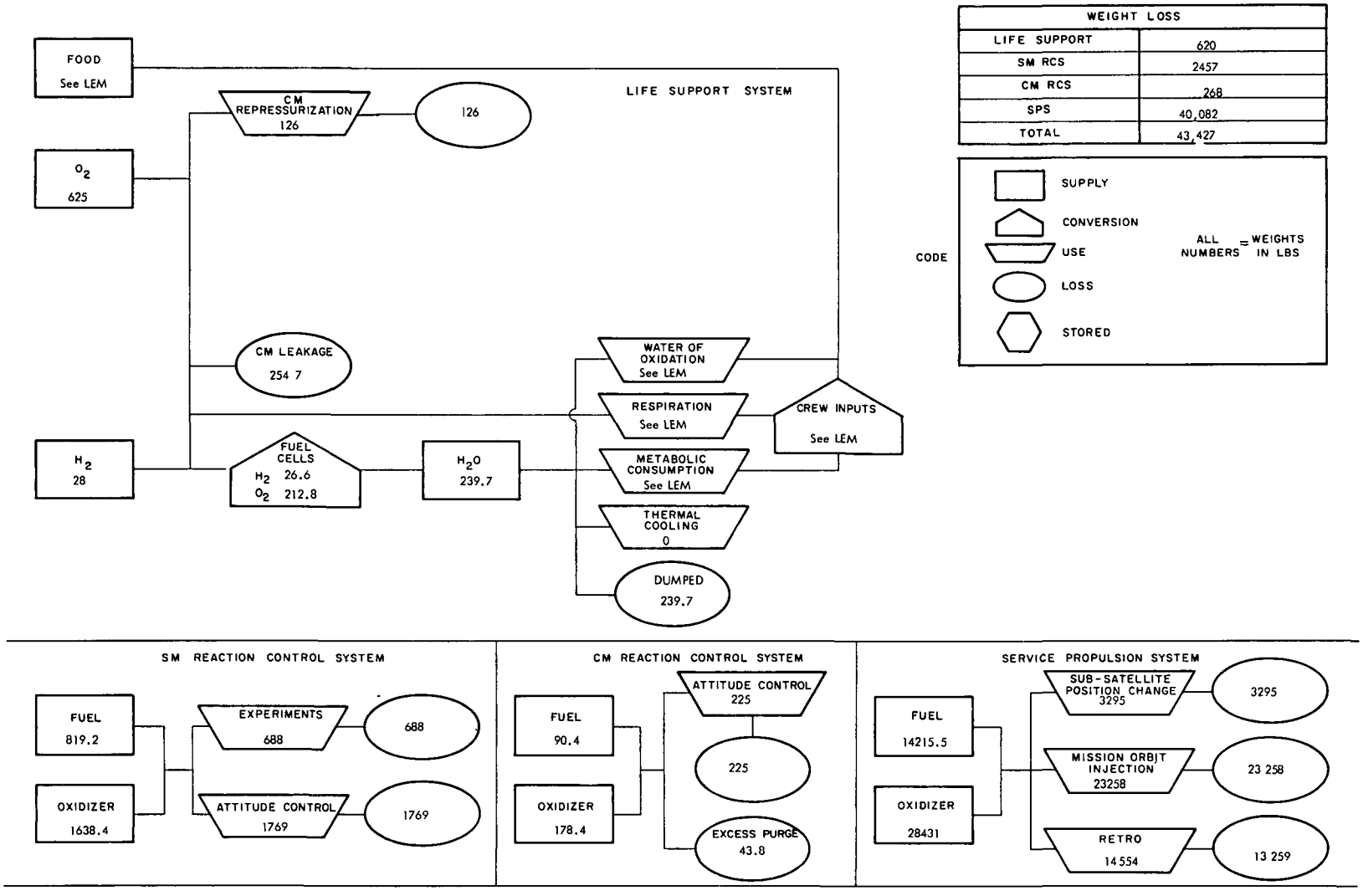


Figure 3-4. Mission 3 - Expendables Weight Summary Chart (Sheet 2 of 2)

k. Thermal Cooling - Cooling water for the LEM is not required (see 32-1-10.16). The same was assumed for the CM.

l. Crew Inputs - The total weight of supplies consumed by the crew and put back into the system is as follows:

(1) Urine - Produced at a rate of 0.11 pounds per man per hour, for a total of 356 pounds, vented overboard.

(2) Feces - Produced at a rate of 0.24 pounds per man per day, for a total of 32.4 pounds, stored on board.

(3) Water Vapor - Expiration and perspiration is emitted at the rate 0.2 pounds per man per hour. The vapor, totaling 648 pounds, is recovered and dumped.

(4) Carbon Dioxide - Human respiration waste, produced at a rate of 2.02 pounds per man per day (see SID 63-1370-2). The total amount, 273 pounds, is stored on board in lithium hydroxide filters.

3.2.2.3.2 SM RCS

The SM RCS fuel tanks per quad (see 45) are loaded to 819 pounds. The SM RCS oxidizer tanks per quad (see 45) are loaded to 1638 pounds.

For attitude control during the experiments, the RCS requires 688 pounds (see 32-1-10.28) of fuel oxidizer mixture in a ratio of one to two. The remaining reactants are available for normal attitude control functions and for:

a. CSM booster separation.

b. Ullage maneuver prior to SPS ignition.

c. Separation of the SM and the CM.

3.2.2.3.3 CM RCS

The two CM RCS fuel tanks (see 80-85) are filled to 90.4 pounds. The two CM RCS oxidizer tanks are filled to 178.4 pounds.

Of the total, 225 pounds of reactants provide:

a. Maneuver capabilities during entry after SM separation.

b. Maneuver capabilities in the event of a high-altitude abort after LES jettison.

c. Three-axis control to stabilize CM motion prior to deployment of the ELS parachutes.

Fuel and oxidizer left in the tanks after parachute deployment and final attitude correction is dumped before touchdown.

3.2.2.3.4 SPS

The SPS fuel tank (see 80 - viii) has a total capacity of 15,435 pounds; the fuel load for this flight is 14,215 pounds. The SPS oxidizer tank has a 31,620-pound capacity; the load for this flight is 28,431 pounds.

There are 3,295 pounds required for the two observation post changes (calculated); 23,258 pounds are required to inject into the 19,329-n. mi. circular orbit (see 111) and 13,529 pounds are required for the retro maneuver (calculated).

3.2.2.3.5 LEM RCS

The four LEM RCS fuel tanks (see 32-1-10.16) have a 406-pound capacity; the load for this flight is 383 pounds. The four LEM RCS oxidizer tanks (see 32-1-10.16) have a 819-pound capacity; the load for this flight is 766 pounds.

The amount of reactant required for LEM attitude control is 115 pounds (see 32-1-10.28).

3.2.2.4 Weight Profile

The vehicle weight changes for the portion of the mission beginning with S-IVB cutoff, insertion into the 100-n. mi. parking orbit, and ending with CM re-entry are shown in Figure 3-5. Data for this chart was taken from Reference 111 from the Weight Summary Chart (Table 3-1) and from the Expendables Weight Summary Charts (Figure 3-4).

The vehicle weight at S-IVB cutoff for earth parking orbit insertion is 288,293 pounds. The first S-IVB transfer orbit thrust reduces the weight to 125,131 pounds. S-IVB staging and jettisoning of the LEM adapter reduces the vehicle weight to 89,592 pounds. After mission orbit injection, the weight of the spacecraft (in the 19,329-n. mi. earth orbit) is 61,441 pounds.

The payload weight required for this mission is 57,699 pounds. The differential between maximum vehicle weight and actual payload weight is 3,742 pounds for this mission and is shown as margin.

There are two post changes during the 45-day mission. Weight loss because of oxygen, water, and urine during the first observation post (10 days) brings the weight down to 56,257 pounds. SPS thrusting at the first post change further reduces the weight to 54,610 pounds.

The 3.8-day transit and 10-day post periods reduce the weight to 52,621 pounds. After second post-change SPS thrusting, the weight is 50,947 pounds. Following the 11.5-day transit and 8.7-day post, the spacecraft weight is 48,061 pounds.

LEM staging brings the CSM weight to 34,062 pounds at the beginning of retrofire. Fuel consumption for retrofiring and SM staging then reduces the weight to 10,554 pounds, which is the CM re-entry weight.

3.2.3 CREW TASK ANALYSIS

3.2.3.1 Time Utilization

Crew tasks performed during Mission 3 fall into two basic categories: flight operations and experiment functions. The crew flight-operations required from liftoff to the synchronous orbit, and from initiation of de-orbit to touchdown are detailed in Appendix D. Flight operations activities required of the crew during the synchronous orbit consist primarily of performing system status checks, IMU alignments, and ground communications. Table 3-4 of this report summarizes these system status checks as adapted from the Apollo design reference mission. A detailed task analysis of the crew activities required for conduct of the experiments is also presented in Appendix D.

Analysis of the flight crew operations, and the sleep, eating, hygiene, and rest requirements for a typical orbital day indicated that approximately 24 man-hours are available in a 24-hour day to perform experiments. This allows a total of approximately 1080 man-hours available for experiments over the 45-day mission. Utilizing the number of experiment sequences and the number of test subjects (see 31, 32, 33) recommended for the assigned experiments with the experiment times as outlined in Appendix D, it was determined that approximately 1588 man-hours would be required to conduct the assigned experiments. Therefore, an experiment priority system was established. The priority classifications scheme and priority numbers are as follows:

a. First priority experiments are those that satisfy primary mission objectives.

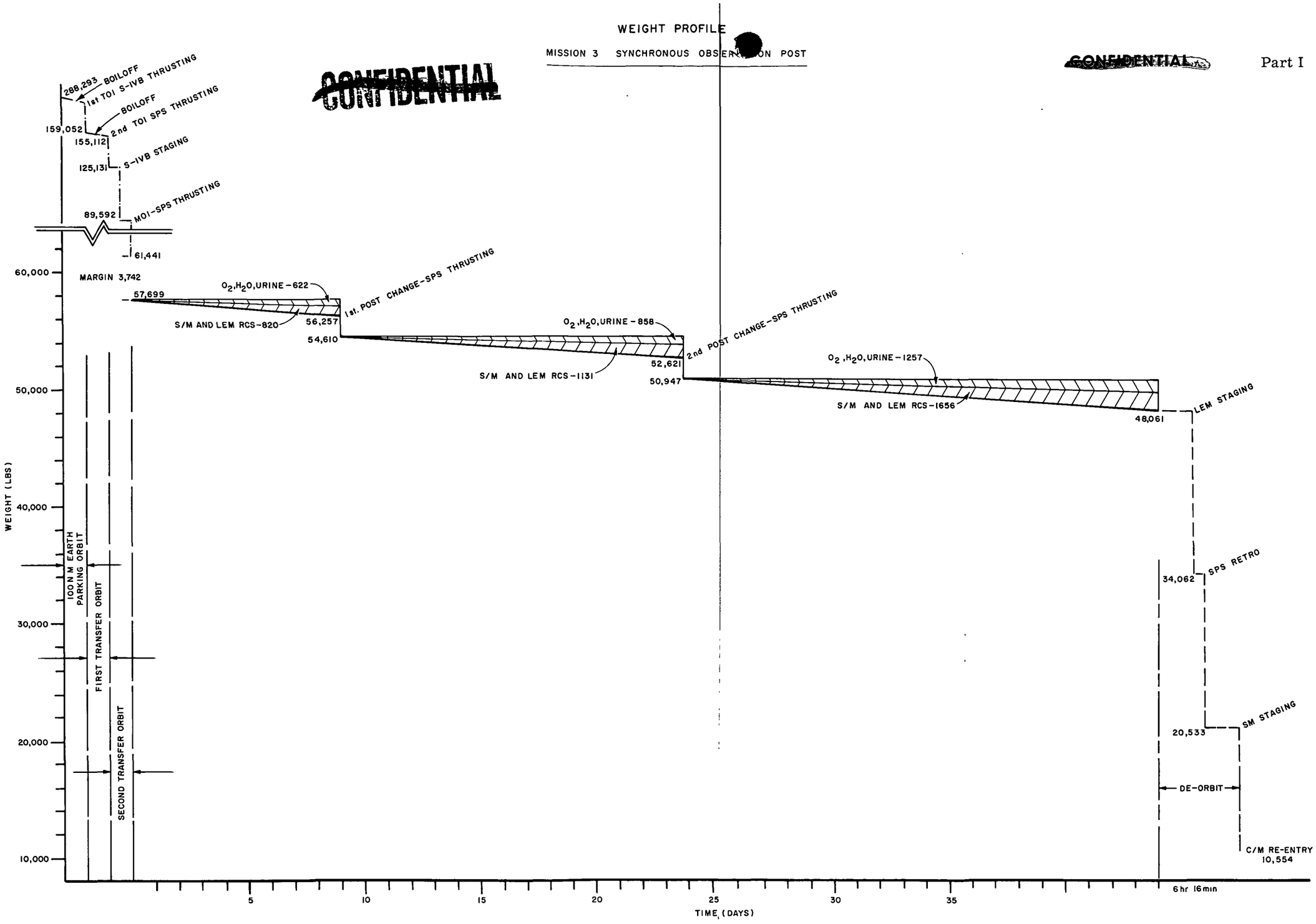


Figure 3-5. Mission 3 - Weights Profile

Table 3-4

Breakdown of Earth Orbital System Status Checks*

Task	Frequency of Performance	Duration (Min)
CM		
Check ECS	Hourly	1
Check EPS	Hourly	3
Check SM-RCS	Every 4 hours	4
Check CM-RCS	Every 12 hours	4
Replace CO_2 and odor absorber	Every 12 hours	8
Charge entry and post landing batteries	Every 6 hours	3
Purge Fuel cells	Every 8 hours	3
Check SPS	Every 4 hours	4
LEM		
Check ECS	Hourly	1
Check PGS	Hourly	1
Check PDS	Hourly	1
Replace CO_2 and odor absorber	Every 12 hours	8
Check LEM-RCS	Every 12 hours	4
4.75 man-hours required for system status checks in a 24-hour day.		

*AMPTF adapted (see 24, 25, 26).

CM tasks taken from translunar and transearth coast phase.

LEM tasks taken from lunar surface phase.

b. Second priority experiments are those that contribute to the assessment of physical health or well being of the astronauts.

c. Third priority experiments are those that satisfy secondary mission objectives.

d. Fourth priority experiments are the biomedical and behavioral experiments that contribute general scientific knowledge.

e. The lowest priority experiments are those that have been performed on previous flights and are of such a nature that omission is not detrimental to the mission objectives.

It should be noted that the priority classification scheme and assignments were arbitrarily made because of the excessive experiment-time load. They do not constitute a recommended or suggested solution to the experiment priority and integration problem.

3.2.3.2 Experiment Priorities

The following are the experiment priorities for Mission 3:

Priorities	Experiment Number and Name	
1	0700 Series	Astronomy - 0701, 0702, 0705B, 0703
1	0801	Conjugate Auroras
2	0501	Radiation Monitoring
2	0100 Series	Biomedical - 0103, 0104, 0105, 0106, 0107, 0109, 0110, 0111, 0112, 0113, 0114, 0117, 0118, 0119, 0121.
3	1001	Subsatellite
3	1300 Series	Extendible Rod - 1303, 1304
3	0400 Series	Microbial - 0401, 0402.
4	0200 Series	Behavioral - 0201, 0202, 0203
4	0100 Series	Biomedical - 0101, 0102, 0108, 0115, 0116, 0120.
5	0406	Spaceborne Microorganisms

The flight operations and experiment scheduling assumptions utilized in scheduling flight 516 are outlined in paragraphs 3.2.3.3 and 3.2.3.4. No communication times for flight operations were programmed on this flight because the communications contact points were not known at the time of scheduling. The actual schedules for the first 14 days of Flight 516 are presented in Figure 3-6. This 14-day sample presents

MISSION NUMBER 3

SYNCHRONOUS OBSERVATION POST

LEGEND

SYSTEM STATUS CHECKS

DAY NUMBER 1

COMMANDER	
NAVIGATOR	
ENGINEER	
HOUR	0 1 2 3 4 5 6 7 8 9 10 11 12 13 14 15 16 17 18 19 20 21 22 23 24
VEH. LIGHT	

DAY NUMBER 2

COMMANDER	Begin Synchronous Orbit	Begin Operations	E/H	Rest	Rest	0501	0117 Ph A	Rest	E/H	Rest	IMU	0108	Rest	0109 and 0110	Sleep	E/H	IMU	Rest	0101					
NAVIGATOR			Rest	Rest	E/H	Rest	0703	Sleep	E/H	0109 and 0110	0705B Phase A, B	E/H	Rest	IMU	Rest	0101								
ENGINEER			Sleep	E/H	0401 A	Rest	0401 B	Rest	0402 ABC Part I	IMU	0108	Rest	E/H	Rest	0701	Rest	Rest	Rest	Rest	E/H	0101			
HOUR	0	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20	21	22	23	24
VEH. LIGHT																									

DAY NUMBER 3

COMMANDER	0117 Ph A	0501	Rest	0401 C	E/H	Rest	IMU	1001 Phase A	E/H	Rest	IMU	0801	Rest	Sleep	E/H	Rest								
NAVIGATOR	Rest	E/H	Rest	0705B Phase C	0705B Phase C	IMU	1001 Phase A	Sleep	E/H	0702 Phase A	Rest	0703	E/H	IMU	Rest									
ENGINEER	Sleep				E/H	Rest	1001 Phase A	0402 ABC Part II	E/H	IMU	E/H	Rest	0702 Phase A	Rest	0118 Method B	Rest	E/H	Rest	IMU	Rest					
HOUR	0	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20	21	22	23	24
VEH. LIGHT																							0402 A Part II		

Figure 3-6. Integrated Mission Schedule (Sheet 1 of 5)

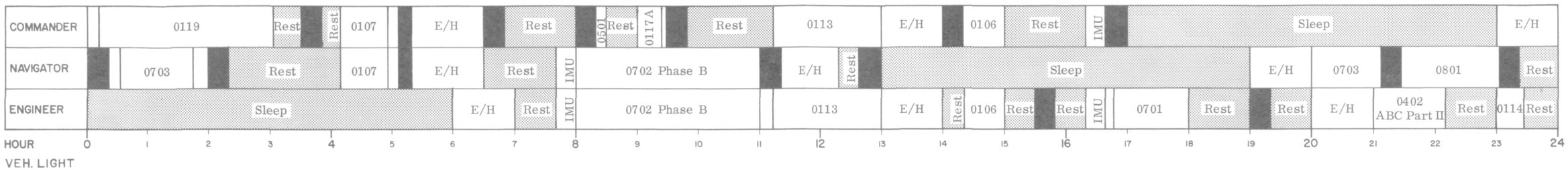
3-41-1

MISSION NUMBER 3

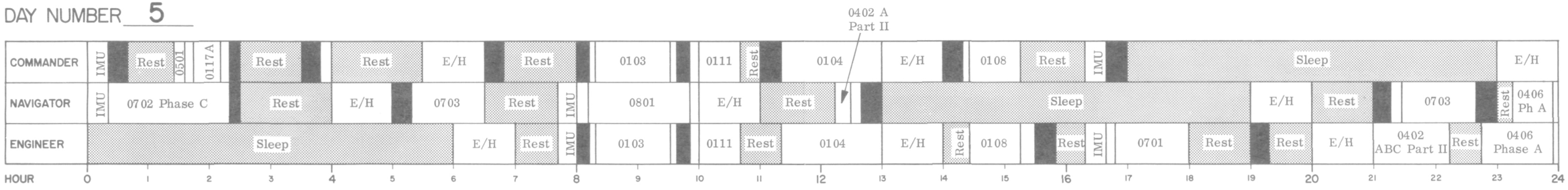
DAY NUMBER 4

SYNCHRONOUS OBSERVATION POST

SYSTEM STATUS CHECKS



DAY NUMBER 5



DAY NUMBER 6

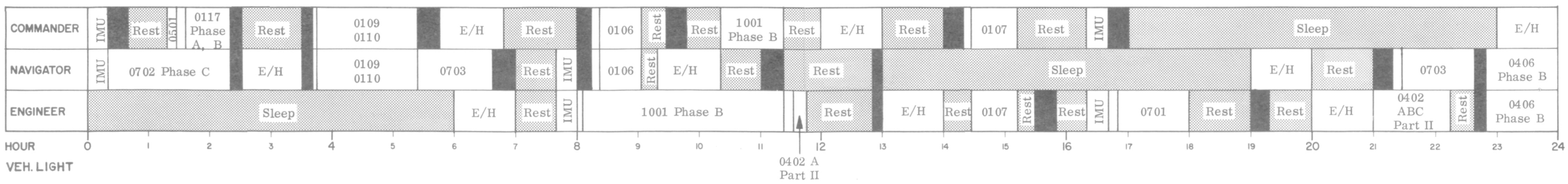


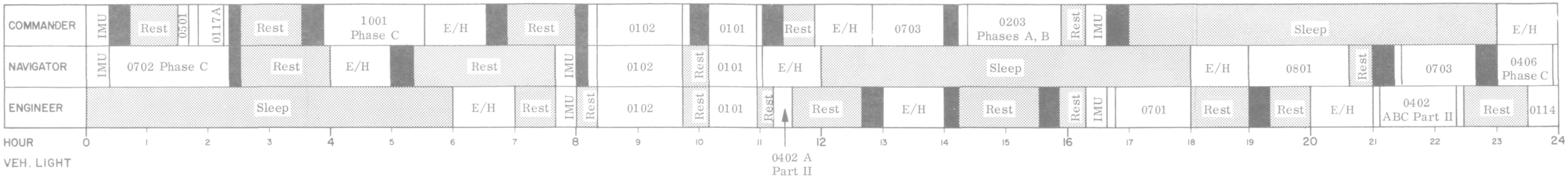
Figure 3-6. Integrated Mission Schedule (Sheet 2 of 5)

MISSION NUMBER 3

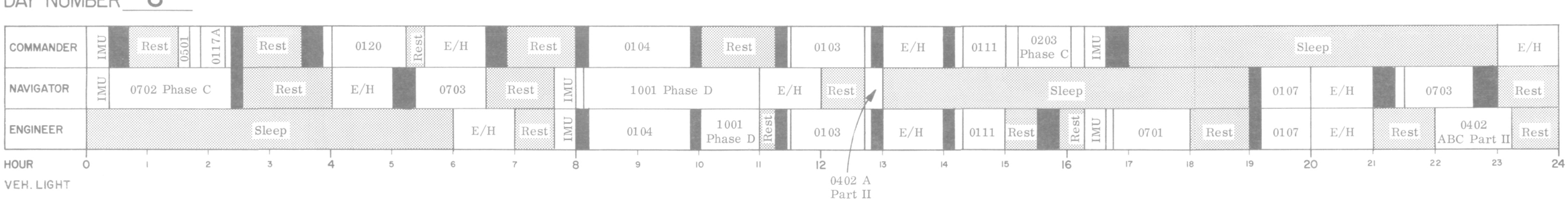
DAY NUMBER 7

SYNCHRONOUS OBSERVATION POST

SYSTEM STATUS CHECKS



DAY NUMBER 8



DAY NUMBER 9

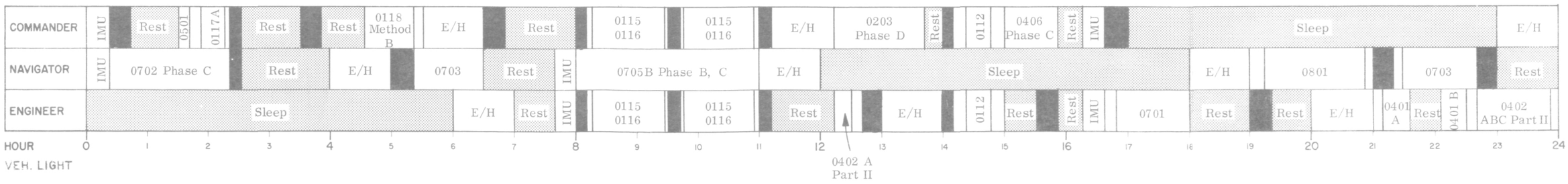


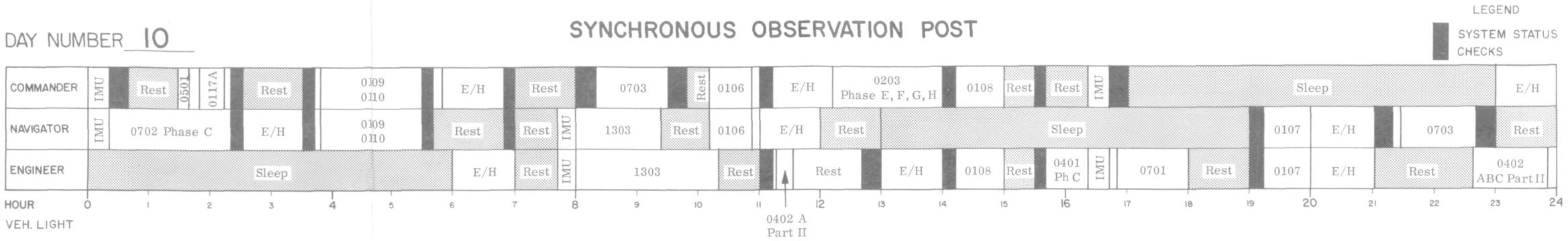
Figure 3-6. Integrated Mission Schedule (Sheet 3 of 5)

3-43-1

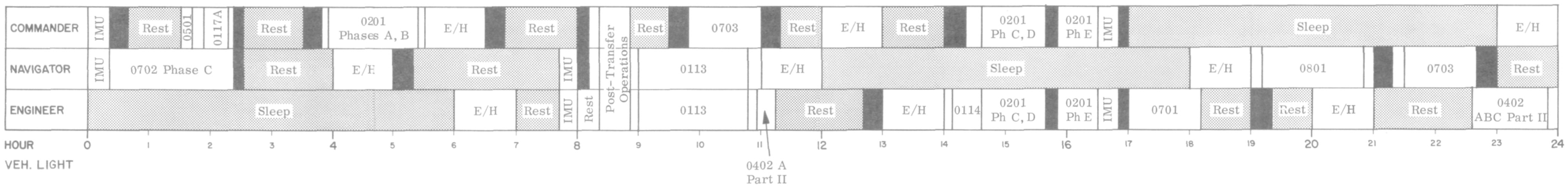
MISSION NUMBER 3

DAY NUMBER 10

SYNCHRONOUS OBSERVATION POST



DAY NUMBER 11



DAY NUMBER 12

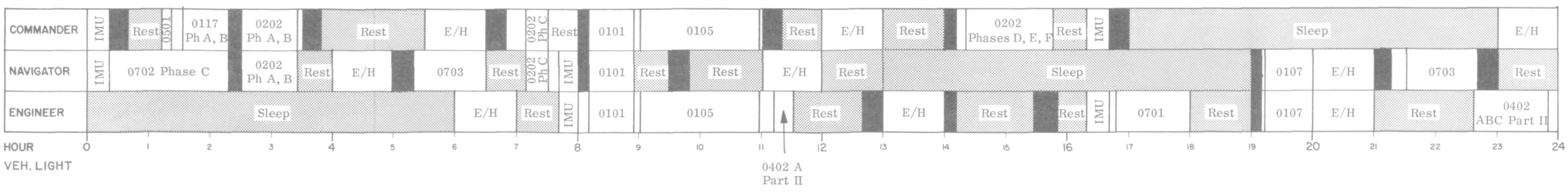


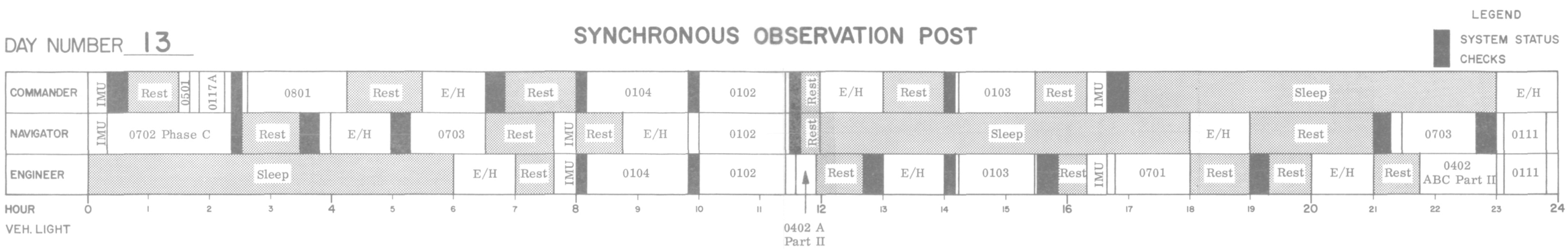
Figure 3-6. Integrated Mission Schedule (Sheet 4 of 5) - Day 11 Operations Evolving Insertion into Drift Orbit are not Shown - Estimated Time 1-Hour (See 24)

3-4-2

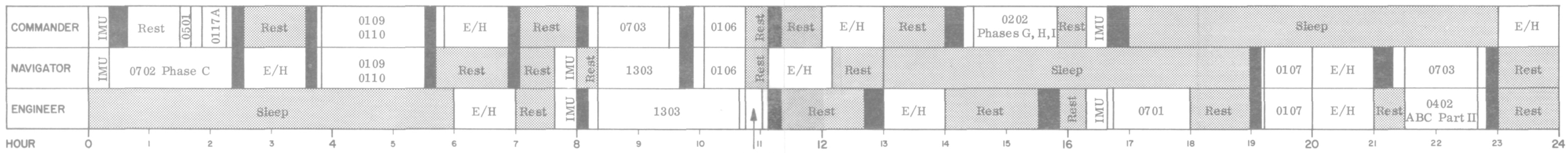
MISSION NUMBER 3

DAY NUMBER 13

SYNCHRONOUS OBSERVATION POST



DAY NUMBER 14



DAY NUMBER _____

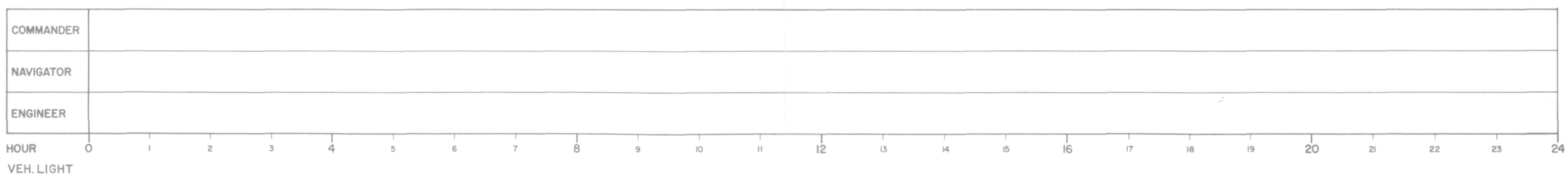


Figure 3-6. Integrated Mission Schedule (Sheet 5 of 5)

5/3-46

3-45.1

the pattern of experiment sequences that can be expected for a complete 45 days. Table 3-5 provides the total amount of time scheduled for each of the experiments, the number of experiment sequences actually scheduled, and the number recommended (31, 32, 33) for a 45-day flight.

The time utilization chart shown below shows the percentage of time actually scheduled for the 4 major crew activities and presents the total man-hours of experiments scheduled. Approximately 53 percent of the recommended experiment time was actually scheduled for the flight.

Mission Time Utilization

Experiments	28.5 percent
Flight Operations	10.5 percent
Personal Time	57.5 percent
Unscheduled Time	3.5 percent
Experiment Time Recommended*	1588 man-hours
Experiment Time Scheduled	834 man-hours

3.2.3.3 Flight Operations Scheduling Assumptions

The following assumptions were considered in studies of flight operations scheduling:

a. Three IMU alignments are required during a 24-hour period (GE).

b. The astronaut day consists of 10-hours on duty and 14-hours off duty (see 24, 25, and 26). The hours off duty will consist of six hours for sleep, five hours for rest, and three hours for eating and personal hygiene.

c. System status checks for flight operations require 4.75 man-hours during a 24-hour period (AMPTF 24, 25, 26 adapted). Any system checks required for experiments are included in the experiment set-up times.

3.2.3.4 Experiment Scheduling Assumptions

For experiment scheduling assumptions, see Experiment Constraints, Volume III, Appendix A, of this report.

*Grumman number of experiment sequences and subjects tested (see 31, 32, 33). Experiment times from Appendix D.

Table 3-5

Mission 3
Summary of Scheduled Experiment Times and Sequences
(14-Day Mission Sample)

Experiment Number	Man-Min Expended (14 Days)	Total Number of Sequences	
		Actual (1)	Recommended (2)
0101	423	3	15
0102	510	2	15
0103	426	3	12
0104	400	3	12
0105	420	1	6
0106	312	4	17
0107	630	6	17
0108	288	3	9
0109	808	4	15
0110			15
0111	240	3	9
0112	50	1	9
0113	428	2	5
0114	75	3	36
0115	240	1	18
0116			4
0117A	365	13 (Phase A)	45 (Phase A)
0117B		2 (Phase B)	6 (Phase B)
0118	144	2	5
0119	170	1	5
0120	70	1	9
0121	0 (3)	0	2
0201	277	1 (Phase A-E)	45 (Phases A-C) 21 (Phases D, E)
0202	309	1	45
0203	331	1 (Phase A-H)	45 (Phases A-C, F-H) 21 (Phases D, E)
0401	142	2	6
0402	1094	24 (Phase A) 12 (Phase B,C)	80 (Phase A) 40 (Phase B, C)

Table 3-5

Mission 3
Summary of Scheduled Experiment Times and Sequences
(14-Day Mission Sample) (Cont.)

Experiment Number	Man-Min Expended (14 Days)	Total Number of Sequences	
		Actual[1]	Recommended[2]
0406	242	2	4
0501	130	13	45
0701	840	12	30
0702	1890	1 (Phase A, B) 10 (Phase C)	1 (Phase A, B) 40 (Phase C)
0703	1656	24	90
0705B	480	2	4
0801	679	7	24
1001	1048	1 (Phase A-D)	1 (Phase A, B)
1303	480	2	6
1304	0[4]	0	14

(1) Actual number of sequences scheduled for 14 days (GE).
(2) Recommended number of sequences for 45 days (38, 39, 40).
(3) Scheduled after day 30.
(4) Scheduled while over second synchronous observation post, days 15 through day 24.

3.2.4 DATA FLOW ANALYSIS

3.2.4.1 General

Data flow characteristics - data recording and transmission, magnetic tape storage and data collection - are detailed in the following paragraphs. The data flow was profiled by using the analysis presented in Part II, Section 3.

3.2.4.2 Data Profiles

The spacecraft during Mission 3 will be in continuous contact with at least one ground telemetry station for all experiment sequences; therefore, it was assumed that no in-flight data recording requirements exist. Profiles for data recording and transmission, and magnetic tape storage, were not generated for this mission.

3.2.4.3 Data Collection Summary

The data collection requirements for Mission 3 are summarized in Table 3-6. The first column designates each experiment by experiment number (see 38, 39, 40, 77). The exception to this is series 0100, the biomedical experiments, and series 0200, behavioral experiments. The next column is the maximum number of special analog channels that must be recorded at any one time because of that experiment. The next indicates whether or not the digital output of a PCM multiplexer must be recorded. The fourth column indicates the minutes of recording time required by each experiment sequence. The fourth and fifth columns indicate the total number of sequences and total minutes of recording time respectively required by that experiment during the course of the mission.

The next column contains a graphically - derived value of the total transmission time available during the mission and is entered only in the over-all totals. The figure given for "Recorded Backlog" is the difference between the over-all total "high rate" PCM and analog data recording time and the over-all total available transmission time. Further experiment data collection details are given in Part II, Section 3.

Several forms of data are required on Mission 3 for which no detailed estimates of weight and volume were available. For experiments in which estimates could be made the data weights total 305 pounds, which exceeds the assumed recoverable payload weight of 250 pounds.

Table 3-6

Mission 3

Data Collection Summary

Experiment	Data Recording							Recovery Requirements				Remarks
	Max. No. of Channels in Use at One Time		Minutes per Sequence	Total No. of Sequences	Total Minutes Recorded	Transmission Time Avail. (Minutes)	Recorded Back Log (Minutes)	Magnetic Tape		Pounds of Film	Miscellaneous	
	Anal.	Dig.						Reels	Lbs.			
	*	*				**	***	**	**			* Assumed time sharing of channels as required. ** Summed only for total mission. ***Difference - Recorded minus Transmitted - only for total mission.
0100	6	1	*	111**	3904					X	X	* Varies for each experiment in series (see data collection charts Part II, Section 3). **Varies for each experiment in series; number (111) is total for series in which recorder is used.
0200	1	1	*	18**	1410					145	X	* Varies for each experiment in series (see data collection charts Part II, Section 3). **Varies for each experiment in series; number (18) is total for series in which recorder is used. X indicates a requirement - no data on rates.

Table 3-6

Data Collection Summary (Cont.)

Mission 3

Experiment	Data Recording							Recovery Requirements				Remarks
	Max. No. of Channels in Use at One Time		Minutes per Sequence	Total No. of Sequences	Total Minutes Recorded	Transmission Time Avail. (Minutes)	Recorded Back Log (Minutes)	Magnetic Tape		Pounds of Film	Miscellaneous	
	Anal.	Dig.						Reels	Lbs.			
0501		1	2	40	80					X		
0401			7	6	42					X	X	
0402			4	40	160					X	X	
0406			10	6	60							
0701	1		30	30	900							
0702	1	1	45	40	1800					20*	100**	* 1000 feet of film. **1600 - 4 x 5 inch glass plate photographs.
0703		1	20	90	1800					6		
0705B	1	1	5	96	480							
0801		1	60*	24	1440							*A few minutes to a few hours for each acquisition - estimated average 60 minutes.
1001	1	1		8	390					15*		*2 to 30 pounds depending on variety and extent of the small maneuverable satellite operations - estimated average 15 pounds. X indicates a requirement - no data on rates.

Table 3-6

Data Collection Summary (Cont.)

Mission 3

Experiment	Data Recording							Recovery Requirements				Remarks
	Max. No. of Channels in Use at One Time		Minutes per Sequence	Total No. of Sequences	Total Minutes Recorded	Transmission Time Avail. (Minutes)	Recorded Back Log (Minutes)	Magnetic Tape		Pounds of Film	Miscellaneous	
	Anal.	Dig.						Reels	Lbs.			
1303		1	121	6	726					9		
1304			30	14	420					10		
Total	6*	1*			13612	Continuous	0	0	0	205+	100+	Total number of channels in use at one time during mission, not a column summation, assumes time sharing as required.

3.2.5 TRAJECTORY ANALYSIS

3.2.5.1 General

Mission 3 utilizes a Saturn V launch vehicle to place an experiment package into an equatorial synchronous orbit. The mission trajectory includes a 100-n. mi. parking orbit, a Hohmann transfer to synchronous altitude, an intermediate elliptic orbit, and a final insertion into synchronous orbit. The mission duration is 45 days, and the hover point is changed twice during the mission for a total of three hover points. Touchdown is in the primary recovery area south of Hawaii, on the equator.

3.2.5.2 Assumptions

The launch time was assumed to be 0700 EST, or 1200 Greenwich mean time. The selection was arbitrarily made. There are no mission constraints other than daylight launching that affect the launch time. Reference 49 was used as the basic document for weights and timing, and modifications were made where appropriate. The major assumption is the third S-IVB burn, used at apogee. A 10 percent ΔV contingency was assumed in all SPS maneuvers.

3.2.5.3 Description

The space vehicle is launched from Cape Kennedy at a launch azimuth of 90 degrees. The S-IC and S-II stages are burned completely, and a partial burn of the S-IVB is used to insert the vehicle into a circular parking orbit at an altitude of 100 n. mi., with an inclination of 28.5 degrees. The launch flight dynamics parameters are shown in Figure 3-7. In parking orbit, various system checks are made and the orbit is verified. After 102 minutes in parking orbit, slightly before the third equatorial crossing, the S-IVB is re-ignited and the vehicle is inserted into a Hohmann transfer orbit with an orbital inclination of 26.5 degrees. (The S-IVB loses 1385 pounds of fuel through boiloff during the parking orbit.) Contact is made with the Ascension island tracking station during the insertion burn and is maintained for 2.4 minutes after engine cutoff. The vehicle coasts to synchronous altitude and, the S-IVB, after having lost 3940 pounds of propellant because of boiloff during the transfer coast, is restarted for a third burn. All the remaining usable S-IVB propellant is burned (604 pounds more than that burned in Reference 49, Table 16) to accomplish a 17.4-degree plane change. The vehicle then coasts for one full orbit in the intermediate orbit, which has an orbital period of 13.70 hours, an inclination of 9.1 degrees, and a perigee altitude of 5294 n. mi. Transposition and docking is performed and the burned-out

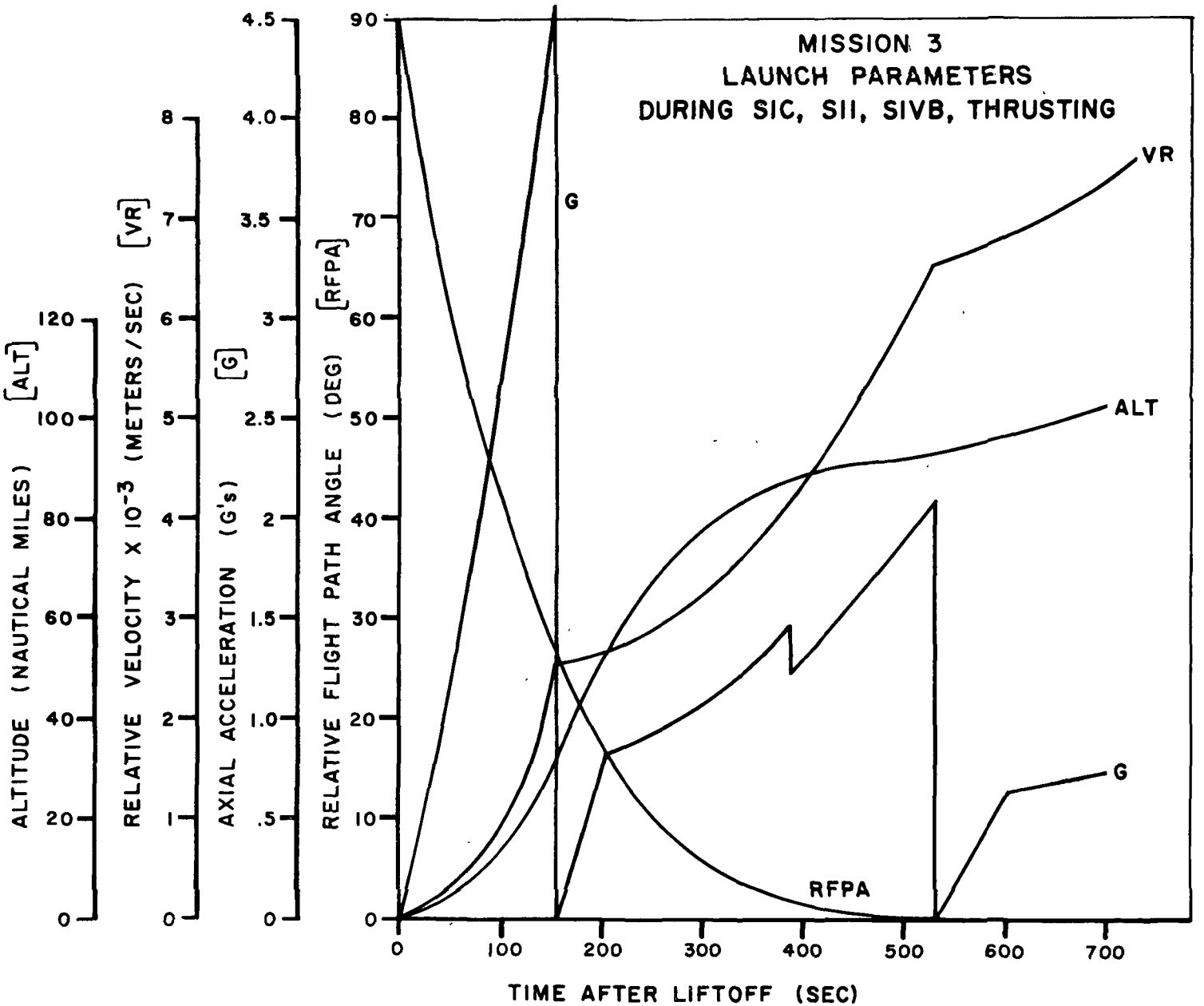


Figure 3-7. Launch Flight Dynamics Parameters

S-IVB stage is jettisoned during this orbit. After completion of the orbit, the SPS is ignited, performs an additional plane change of 9.1 degrees, and completes insertion into the equatorial synchronous orbit. The result of this sequence of events is that the vehicle hovers over a point on the earth at 0 degrees latitude and 126 degrees west longitude. This sequence is illustrated by the diagram shown in Figure 3-8.

After approximately 10 days, the SPS is ignited and a deboost velocity increment of 130 feet per second is implemented. The resulting orbit has a perigee of 18,213 n. mi., after four orbits (92 hours) the vehicle has drifted eastward to a point over the earth's surface 56 degrees west of the first hover point at 70 degrees west longitude. The SPS is then re-ignited and a velocity increment is added to place the vehicle again in synchronous orbit. After approximately 10 days at this hover point, the procedure is repeated and the third and final hover point is established at 90 degrees east longitude (see Phase 6, Mission Sequence). The velocity increments and transit time computations were based on the usage of all available SPS fuel. The first hover point will be the first available nodal crossing after parking orbit crew tasks are completed. The third hover point is dictated by re-entry considerations, ranging from 113.5 to 130 degrees west of the Hawaii primary recovery area. The second hover point was chosen to lie south of the north geomagnetic pole located at 70 degrees west and 79 degrees north. (The south geomagnetic pole lies at 110 degrees east and 79 degrees south.) The drift orbit parameters are shown by the graph presented in Figure 3-9. The vehicle will be in sunlight throughout the mission once synchronous orbit is achieved, and continuous tracking will be available from at least one ground station.

On the 45th mission day, the SPS is ignited once again and performs the deboost maneuver. After the burn the service module is jettisoned and the command module begins the return coast. From the third hover point, the touchdown point will be in the primary recovery area southwest of Hawaii. No plane changes are made on the return trajectory or during re-entry, and touchdown occurs on the equator. The re-entry phase is based on a 26-n. mi. (see 26, 29) vacuum perigee. Figure 3-10 gives the nominal profile. The nominal footprint is then 113.5 to 130 degrees of earth central angle. These values were used to locate the touchdown points from the three hover point stations on the geographic map.

A detailed listing of flight dynamics parameters for all powered phases of the mission is given in Table 3-7.

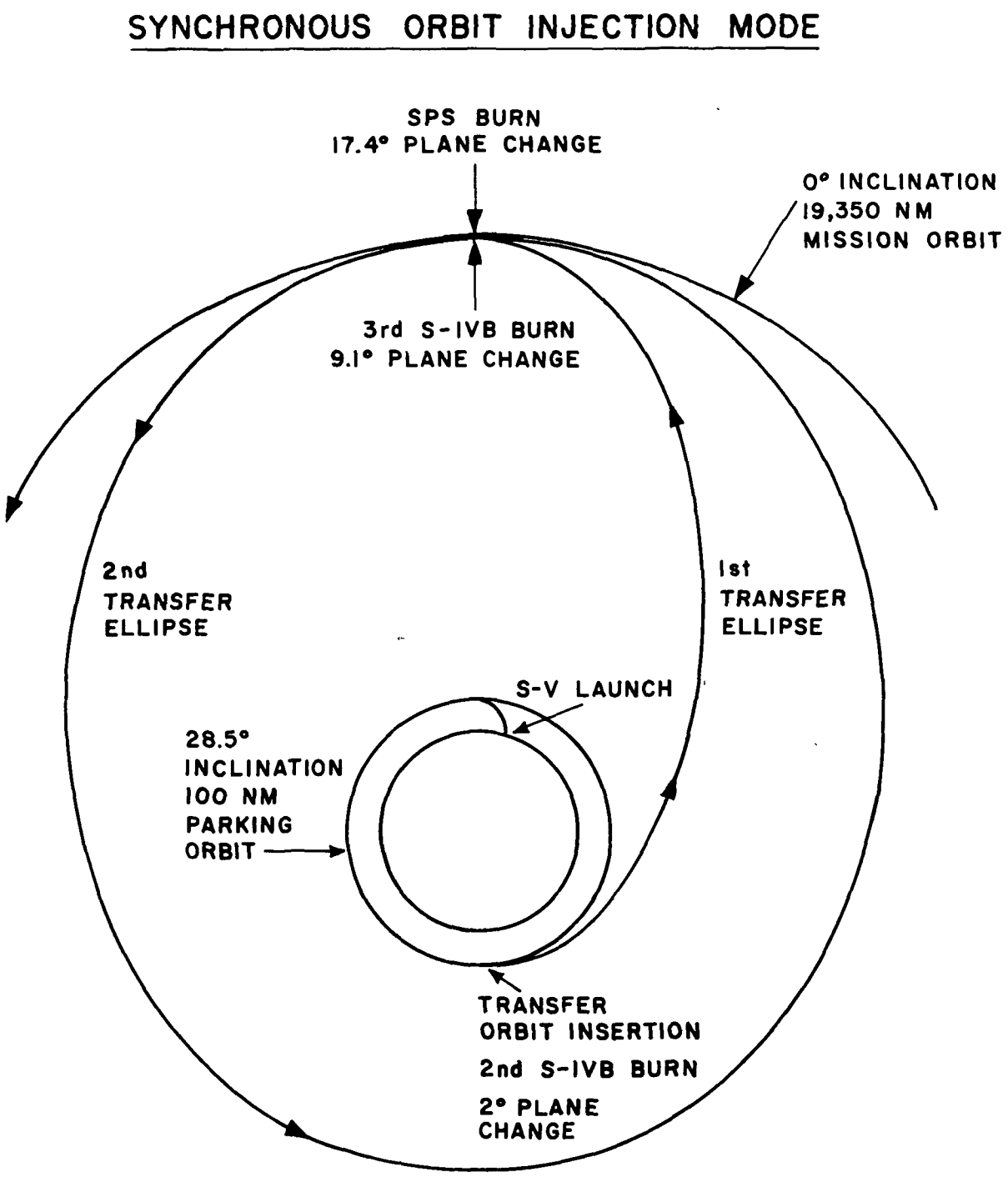


Figure 3-8. Synchronous Orbit Injection Mode

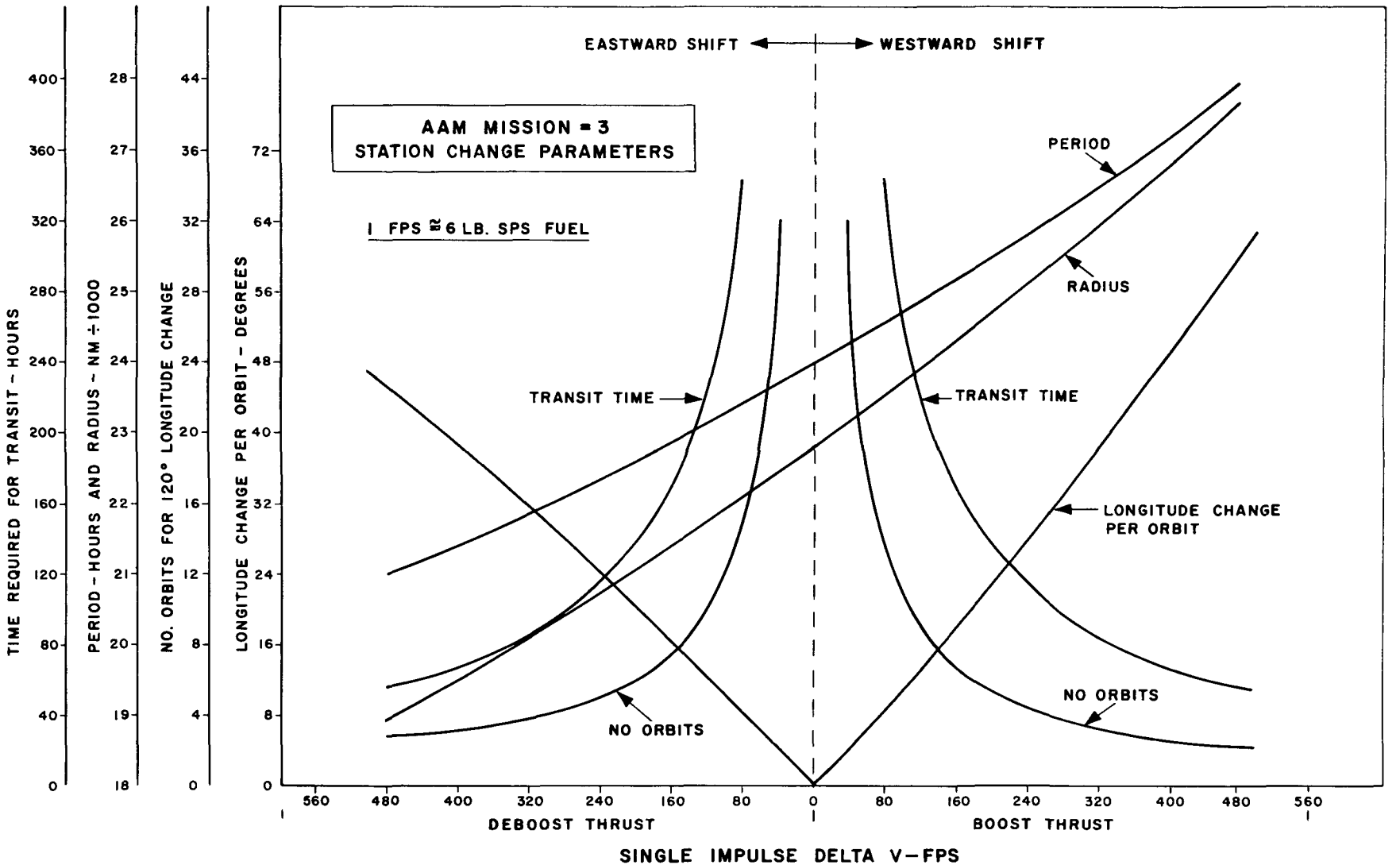


Figure 3-9. Drift Orbit Parameters

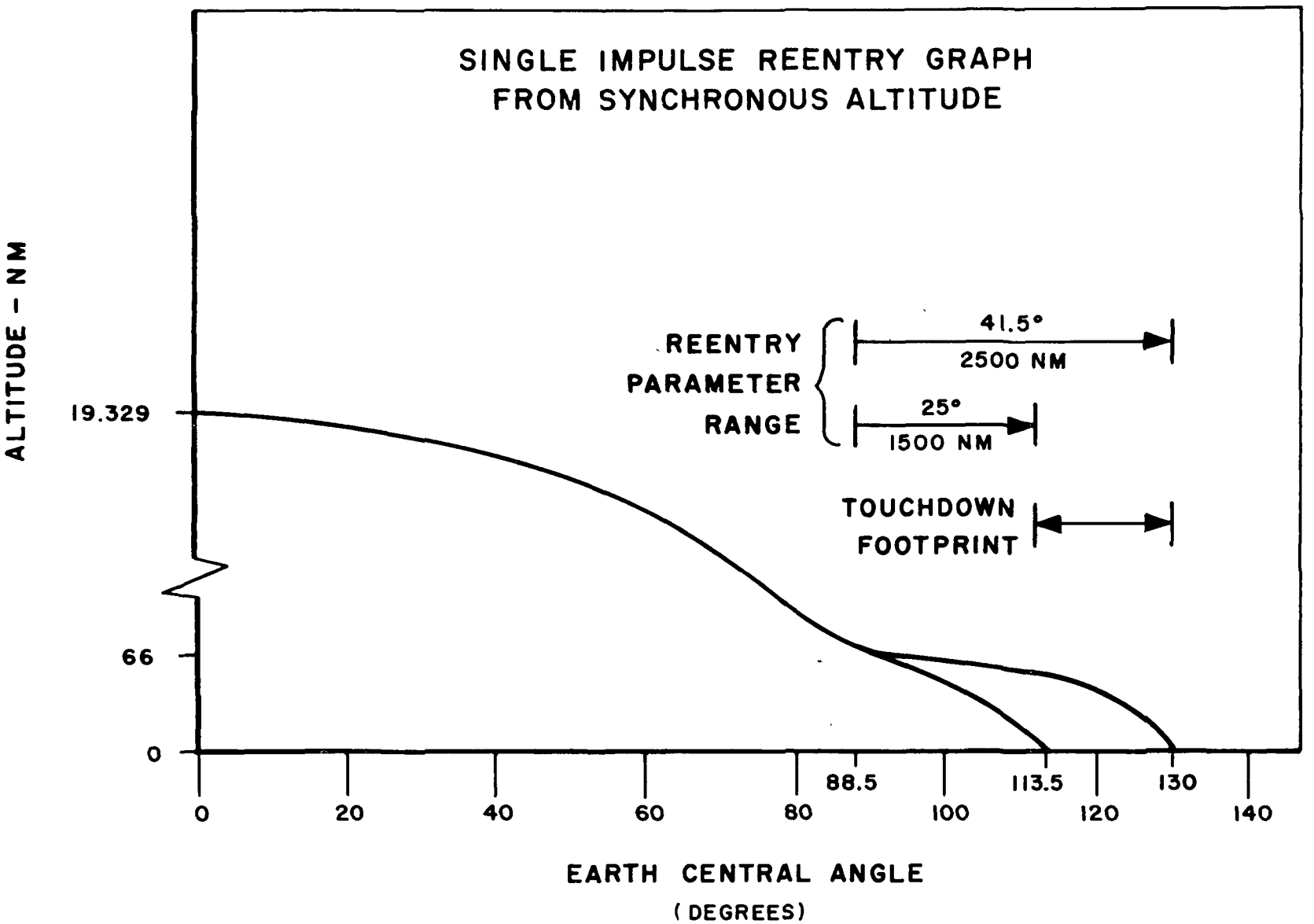


Figure 3-10. Single Impulse Re-entry Graph from Synchronous Altitude

Table 3-7

Saturn V Reference Trajectories, Synchronous Orbits (i = 0 Degrees)

Trajectory Event	Time (seconds)	Geodetic Latitude (degrees)	Longitude (degrees)	Velocity[1] (feet per second)	Altitude (feet)	Flight[1] Path Angle (degrees)	Azimuth[2] (degrees)	Vehicle Weight (pounds)
1. Liftoff	0.00	28.65	-80.64	1,340	0	0.00	-	6,341,035
2. End Vertical Rise, Begin Zero-lift Flight	12.00	28.65	-80.64	1,343	517	3.87	90.0	5,994,575
3. Shutdown of S-IC Inboard Engine	154.57	28.64	-79.84	8,693	192,289	20.01	90.7	1,878,416
4. Shutdown of S-IC Outboard Engines, Begin Coast	158.57	28.64	-79.75	9,141	204,345	19.52	90.8	1,786,027
5. Jettison of S-IC, S-II Ignition, Begin Pitch-up Maneuver[3], Mixture Ratio = 5.0	162.37	28.64	-79.67	9,101	215,750	18.90	90.8	1,404,382[4]
6. End Pitch-up Maneuver, Change Mixture Ratio to 5.4	172.37	28.64	-79.44	9,238	244,342	17.50	91.0	1,380,086
7. Jettison S-IC/S-II Interstage Adapter Section	188.57	28.63	-79.06	9,514	287,565	15.61	91.2	1,326,876[5]
8. Jettison Launch Escape System	193.57	28.63	-78.94	9,605	300,198	15.05	91.3	1,305,268[6]
9. Change Mixture Ratio to 4.7	386.28	28.32	-72.70	15,439	574,130	2.14	94.9	788,508
10. Shutdown of S-II, Begin Coast	534.19	27.49	-65.04	22,731	606,632	0.44	98.9	456,420
11. Jettison S-II, S-IVB Ignition	538.99	27.45	-64.73	22,730	607,358	0.36	99.1	355,499[7]
12. Burnout of S-IVB, End Integrated Trajectory, Coast to Equator	677.30	25.79	-55.57	25,582	607,831	0.00	103.6	288,293
13. Re-start of S-IVB for Perigee Overspeed (And 2-Degree Plane Change)	6,923.28	3.24	-25.65	25,582	607,831	0.00	118.5	286,908
14. Burnout of S-IVB, Coast to Apogee	7,187.44	-6.17	-8.52	33,657	607,831	0.00	116.5	158,547
15. Re-start of S-IVB for Apogee Maneuver	26,107.36	0.00	80.62	5,234	19,329	0.00	63.5	154,607

Table 3-7

Saturn V Reference Trajectories, Synchronous Orbits (i = 0 Degrees) (Cont.)

Trajectory Event	Time (seconds)	Geodetic Latitude (degrees)	Longitude (degrees)	Velocity[1] (feet per second)	Altitude (feet)	Flight[1] Path Angle (degrees)	Azimuth[2] (degrees)	Vehicle Weight (pounds)
16. Burnout of S-IVB	26,169.26	0.00	80.51	7,514	19,329	0.00	80.9	124,527
17. Service Module Ignition	75,489.26	0.00	-126.45	7,514	19,329	0.00	80.9	84,592[13]
18. Service Module Burnout (Circular Orbit)	75,795.16	0.00	-126.63	10,089	19,329	0.00	90.0	61,441[14]
18a. Completion of Third Hover Point Transit	-	0.00	Various[15]					58,146
19. Service Module Ignition for Deboost Maneuver[13]	0.00	0.00	89.00	10,089	19,329	0.00	90.0	35,119
20. Service Module Burnout	209.00	0.00	89.00	5,219	19,329	0.00	90.0	20,565
21. Command Module Re-entry	18,668.50	0.00	177.5	33,840	66	-6.4	90.0	10,303

(1) Inertial Quantities.
(2) Relative Quantities.
(3) Pitch-up of one-degree per second for 10-seconds approximates optimum trajectory, after which a pitch-down rate of 0.1021 degree per second is maintained to first S-IVB burnout.
(4) After Jettison Event of 381,645 pounds.
(5) After Jettison Event of 9,770 pounds.
(6) After Jettison Event of 8,200 pounds.
(7) After Jettison Event of 100,921 pounds.
(8) 1385 pounds Boil-off Due to Boil-off During 1.2 Parking Orbits to Third Nodal Crossing.
(9) Nautical Miles.
(10) 3940 pounds lost due to Boil-off during coast to apogee.
(11) The difference between 18 and 19 represents the payload left in orbit or consumed during stay.
(12) Allow one orbit prior to S/M operation: Period = 13.70 hours, Perigee altitude = 5294 nautical miles.
(13) After Jettison of S-IVB Event of 39,935 pounds.
(14) Each S/M operation provides for an additional 10 percent velocity contigency over velocity required in weight tables.
(15) Hover Points: (1) -126° (2) -70° (3) 90°

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Data for the launch from liftoff through injection into the 100-n. mi. circular orbit is taken from Reference 49. Computation by General Electric has supplied the remainder of the data. The launch ground trace and hover point locations for this mission are shown on a uniform grid projection in Figure 3-11. The re-entry ground traces and abort landing areas are shown on a uniform grid projection in Figure 3-12. Ground tracking coverage and illumination for the ascent phases is given in Figure 3-13.

3.2.6 ABORT ANALYSIS

3.2.6.1 General

Four major abort modes are applicable to Mission 3 (see 24, 25, 26, 105, 106, 107):

a. LES abort.
b. Suborbital abort.
c. Abort to orbit.
d. De-orbit abort.

The relationship between the mission phases and the abort mode available at any particular point within the phase is shown in Figure 3-14. Ground rules and mode descriptions for each abort mode are presented in the following paragraphs.

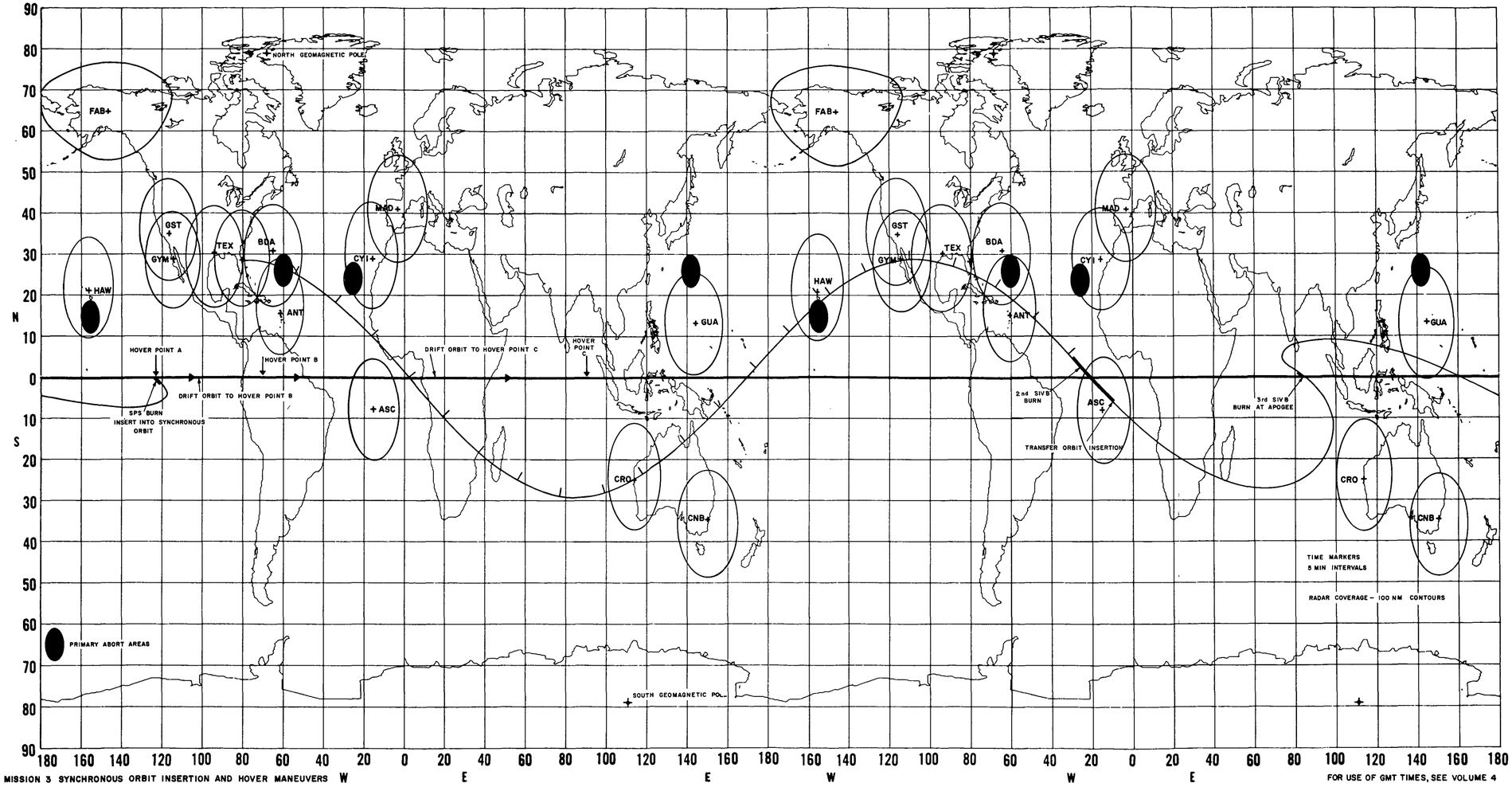
3.2.6.1 Launch Escape System Aborts

3.2.6.1.1 Ground Rules

A LES abort can be initiated anytime between LES enablement during prelaunch and LES jettison during the second stage thrusting period.

LES is the primary abort mode. Launch vehicle engine shutdown must precede LES ignition for all aborts except those initiated during the first 40 seconds of flight. During the first 40 seconds, engine shutdown is inhibited by an AMR range safety constraint.

LES aborts initiated below 120,000 feet are automatically sequenced. In aborts initiated above 120,000 feet the astronauts must jettison the LES manually, after escape motor thrust tail-off, and must orient the spacecraft to a heat-shield-forward attitude.



MISSION 3 SYNCHRONOUS ORBIT INSERTION AND HOVER MANEUVERS

FOR USE OF GMT TIMES, SEE VOLUME 4

Figure 3-11. Launch and Insertion into Mission Orbit Ground Trace

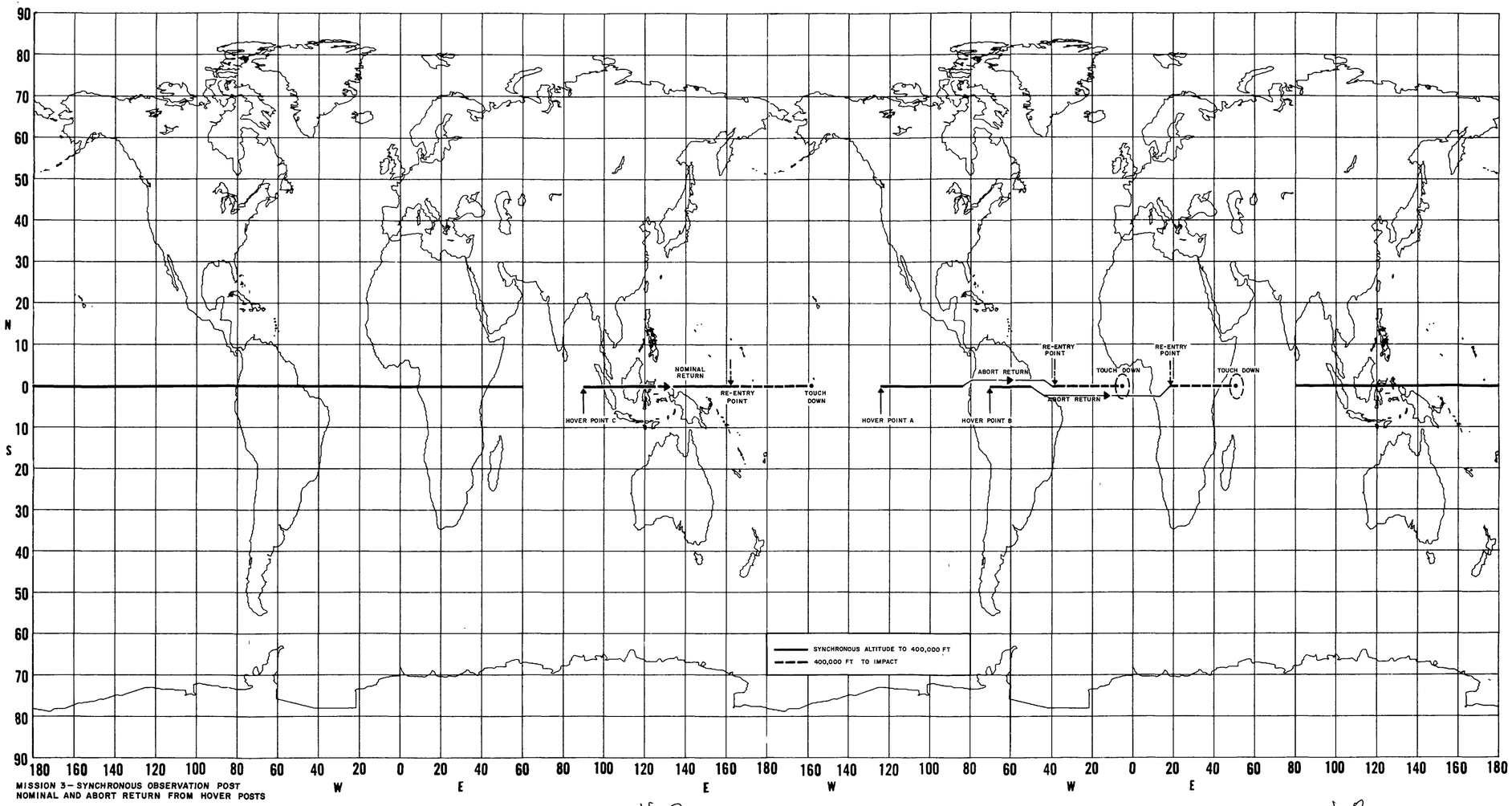


Figure 3-12. Re-entry Ground Trace

MISSION 3 - SYNCHRONOUS OBSERVATION POST
LAUNCH AND PARKING ORBIT

ILLUMINATION AND RADAR COVERAGE CHART

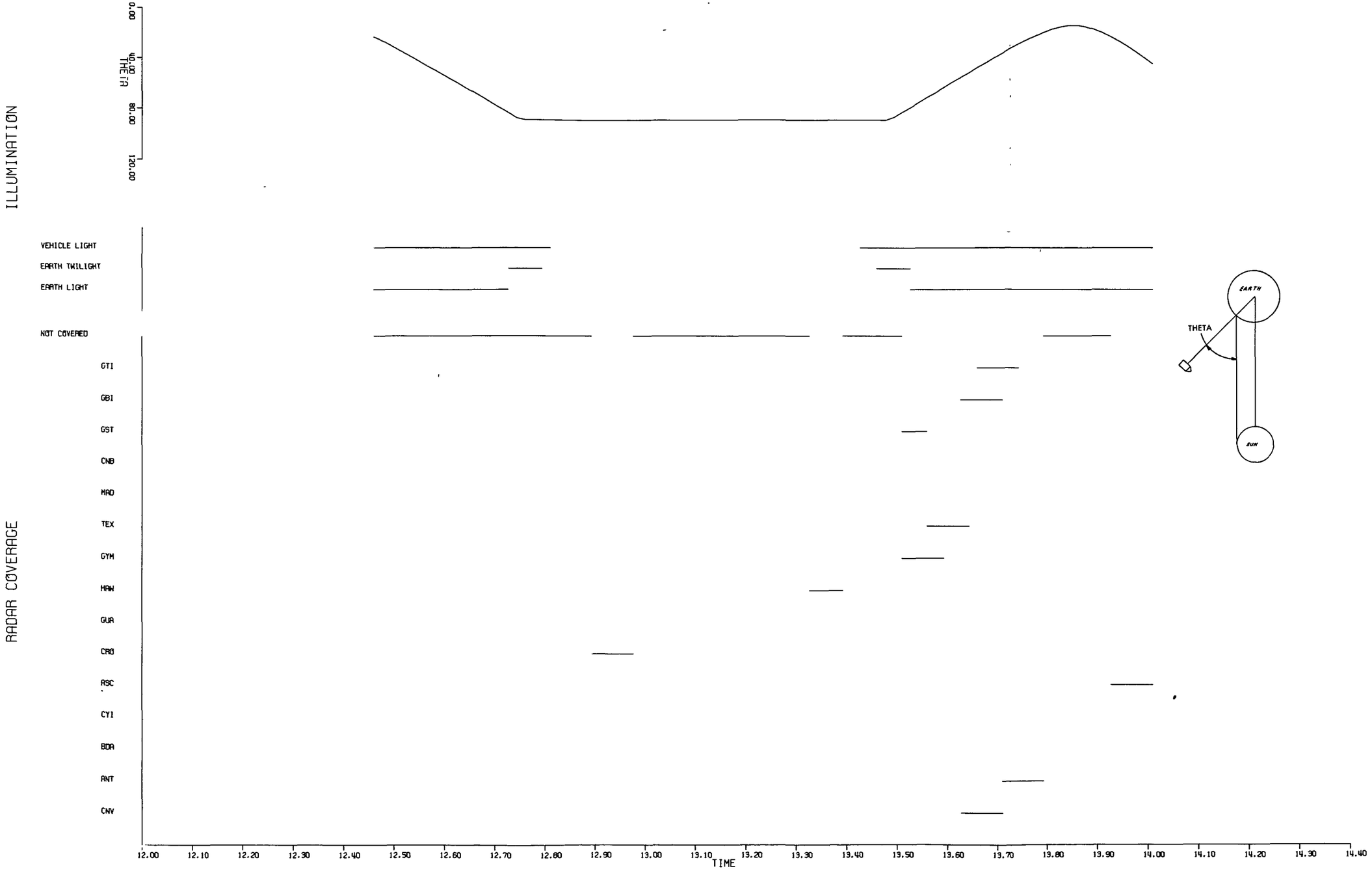


Figure 3-13. Illumination and Radar Coverage (Sheet 1 of 2)

6-65

Part I

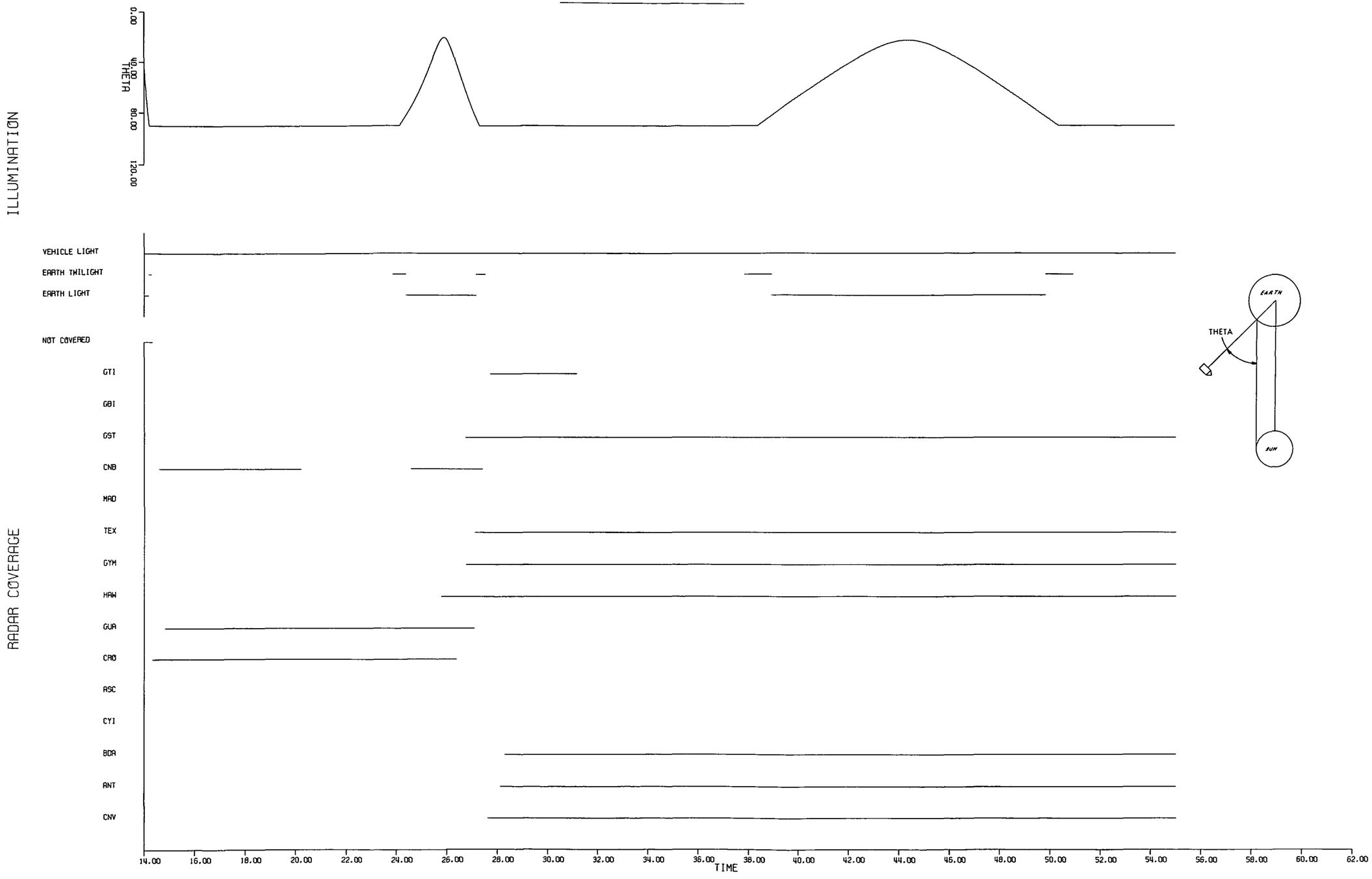
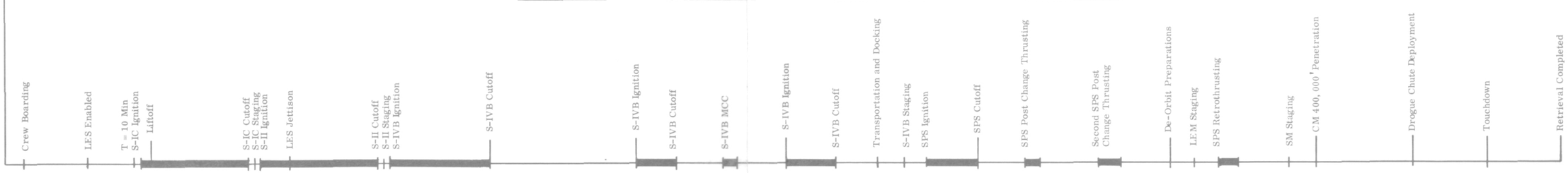


Figure 3-13. Illumination and Radar Coverage (Sheet 2 of 2)

3-66-2

Phase Number	1	2	3	4	5	6	7	8	9	10
Phase Name	Preflight	Earth Ascent	100-N. Mi. Earth Parking Orbit	First Transfer Orbit Coast	Second Transfer Orbit Coast	Synchronous Orbit Coast	De-Orbit	Re-Entry	Parachute Descent	Recovery and Retrieval

Events



Abort Modes:

N/A	Escape Via Launcher Access Arms	LES Abort	Suborbital	Abort to Orbit	SPS De-Orbit	S-IVB De-Orbit	SPS De-Orbit	SPS MCC
N/A		Escape Via Launcher Access Arms	Abort to Orbit	Suborbital	RCS De-Orbit	SPS De-Orbit		SM RCS MCC

Figure 3-14. Mission Phase vs Abort Mode

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3.2.6.1.2 Mode Description

The operational events following the initiation of a LES abort, automatically controlled by sequential timers and barometric switches are shown in Figure 3-15. At abort initiation, launch vehicle engine shutdown is commanded (except during the first 40-second range safety lockout). The launch escape and pitch control motors are then fired simultaneously. The pitch motor burns for approximately 0.5 second to provide lateral separation from the booster, while the escape motor thrusts for about 3.5 seconds to provide longitudinal separation.

At 11 seconds after abort, the canards are deployed. At 14 seconds after abort, if the 25,000-foot altitude barometric switch (closed during prelaunch) is still closed, the tower jettison motor separates the LEM from the CM including the CM apex cover. If the switch is open, LES jettison is delayed until reaching 25,000 feet altitude on descent, when the barometric switch is closed again. Drogue parachute deployment occurs two seconds after LES jettison. Main parachute deployment is controlled by a timer and barometric switch similar to that of the LES jettison. At 12 seconds after drogue deployment, if the 12,000-foot barometric switch is still closed, the drogue chutes are disconnected and the main chutes are deployed. If the switch is open, the drogue chutes are retained until reaching 12,000 feet on descent, when the switch is closed again.

For aborts above approximately 120,000 feet, the canard subsystem is ineffective and the pilot must manually orient the CM in a heat-shield-forward attitude. To do so, the LES must be manually jettisoned by the pilot, after launch escape motor thrust tail-off (about 8 seconds). The CM apex cover is left on. Automatic RCS rate damping begins at tower jettison. Upon reaching 25,000 feet on descent, the apex cover is jettisoned upon closing of the previously discussed barometric switch. Two seconds later the drogue parachutes are deployed. At 12,000 feet another barometric switch closes, disconnecting the drogue chutes and deploying the main chutes.

3.2.6.2 Suborbital Aborts

A suborbital abort can be initiated anytime between LES jettison and earth orbit insertion.

3.2.6.2.1 Ground Rules

A suborbital abort is the primary abort mode from LES jettison during the S-II thrusting period to the ignition of the Saturn S-IVB stage. A suborbital abort is the secondary

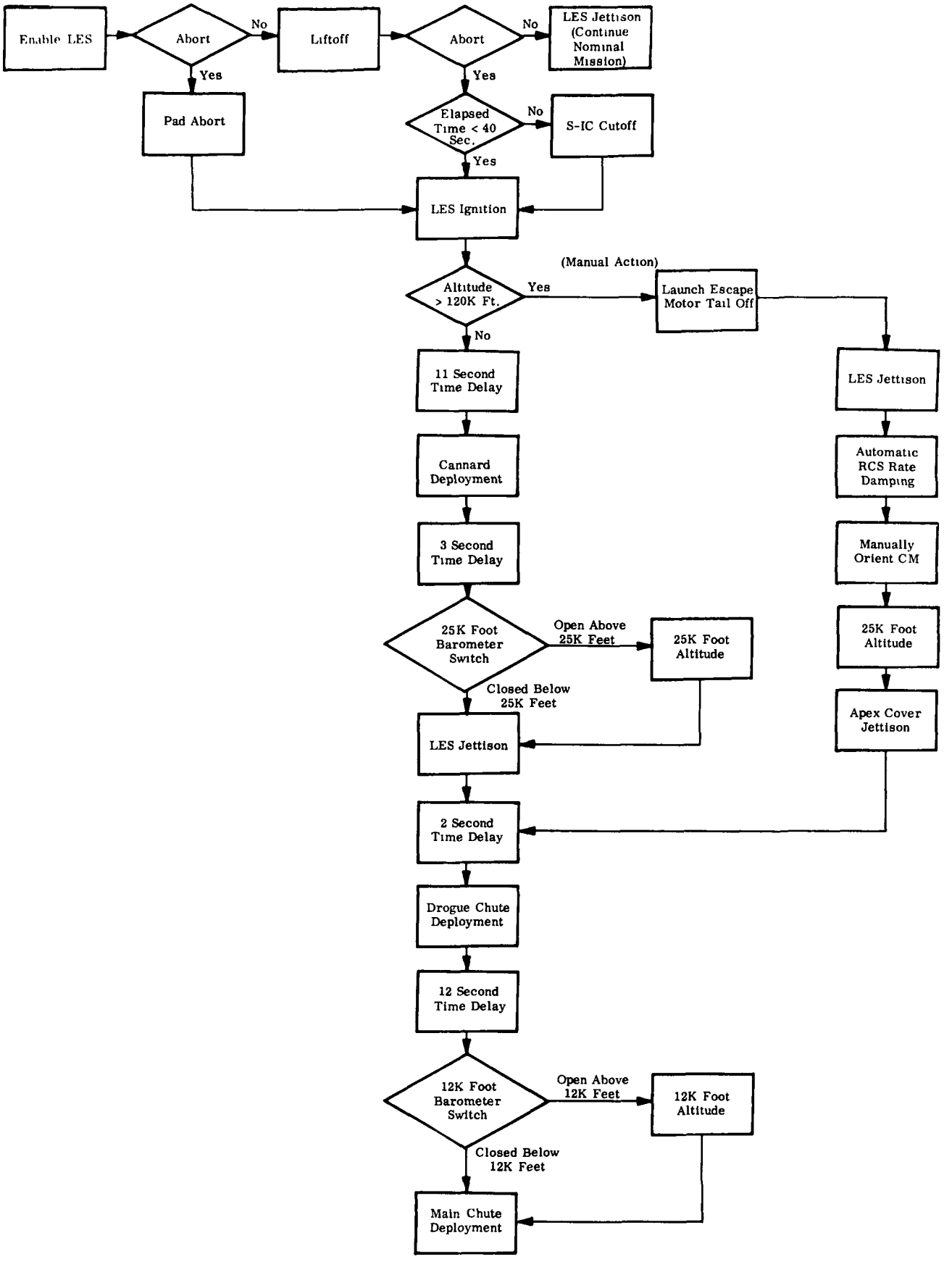

Figure 3-15. LES Abort

abort mode from S-IVB ignition to the completion of the earth orbit insertion maneuver (S-IVB cutoff). In the primary abort mode the CM lands in the AMR; in aborts in the secondary mode the CM lands in the Indian Ocean recovery area.

Launch Vehicle shutdown must precede initiation of spacecraft abort action. The CSM abort separation procedure utilizes both a SM RCS and SPS burn of fixed initiation time, ΔV, and attitude. The SM must be separated from the CM at some fixed increment of time after SM engine shutdown following completion of the suborbital abort maneuvers.

Suborbital aborts before S-IVB ignition (primary abort mode) require that the CM fly a maximum lift trajectory during entry to avoid excessive decelerations. Suborbital aborts after S-IVB ignition (secondary abort mode) require SPS posigrade thrust to obtain the range of the landing site. A constant bank angle of approximately 60 degrees will be flown by the CM during entry.

3.2.6.2.2 Mode Description

The sequence of events during a suborbital abort is shown in Figure 3-16.

Following booster shutdown, which must precede the abort action, the SM RCS is used to provide approximately 45 inches of separation between the CSM and launch vehicle to clear the SPS engine nozzle of the SLA. The nozzle must be clear as soon as possible to damp out any existing pitch and yaw rates that could exist at abort as a result of launch vehicle control system failure. After any necessary damping maneuvers are completed, a turnaround to the required thrust attitude is achieved by pulsing the SM RCS.

Depending on the range control required, the SPS can be burned for variable lengths of time in either a posigrade or retrograde attitude. The time delay to thrust initiation can also be varied. After the CM is separated from the SM, the CM is capable of re-entering ballistically or with variable lift.

SPS range control thrusting is not used for aborts initiated during the period between LES jettison and S-IVB ignition. The CM, flying a maximum lift trajectory, will land within the confines of the Atlantic Missile Range. Therefore, the procedure for the primary abort mode is to use the SM RCS to establish the correct CM/SM separation attitude after the SM RCS launch vehicle separation and SPS damping maneuvers are

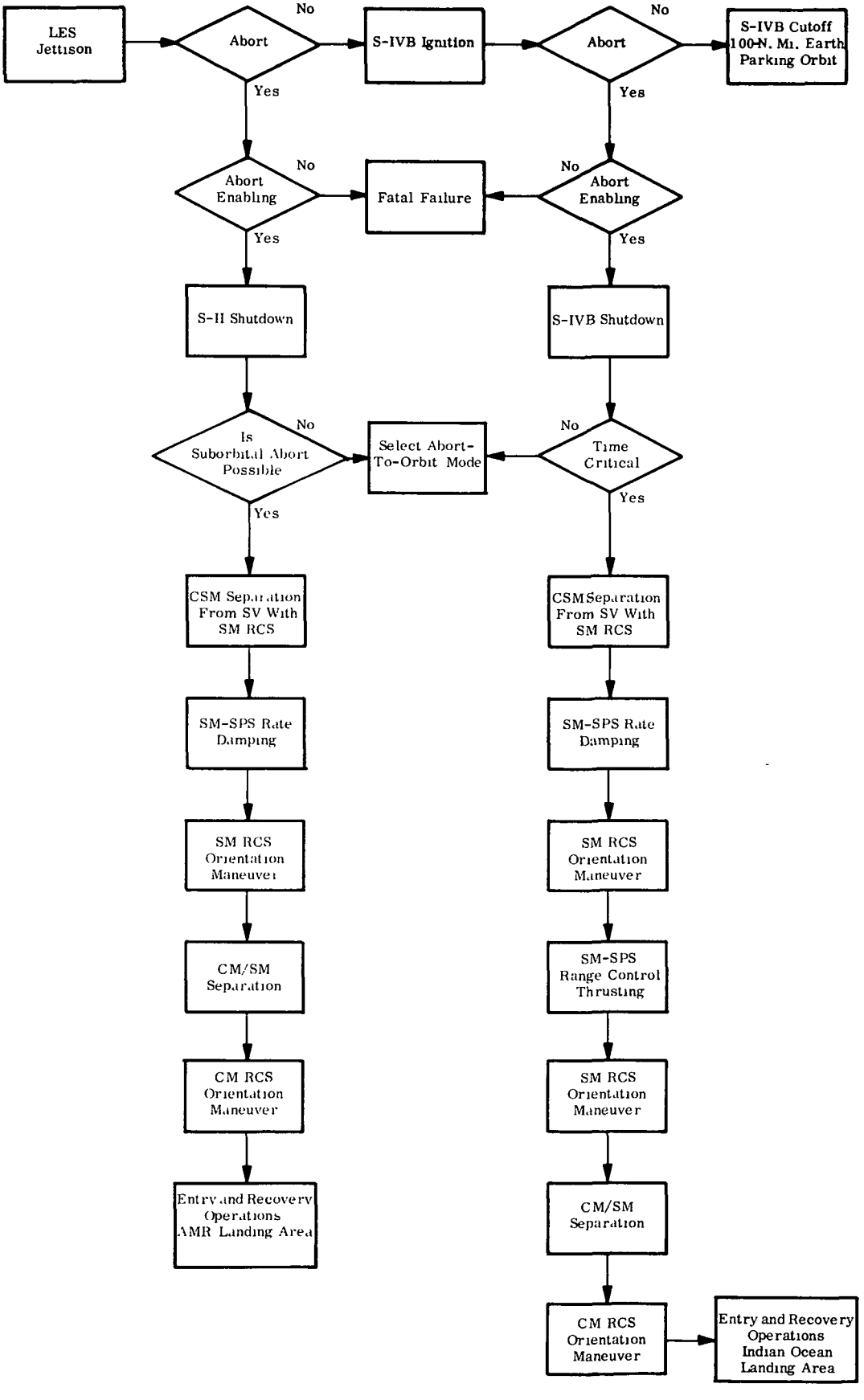

Figure 3-16. Suborbital Abort

completed. The SM RCS must provide the separation velocity. The CM has no translatory thrust capability. Following separation, the CM RCS is used to orient the CM for a maximum-lift entry.

During the period from S-IVB ignition to S-IVB cutoff (earth orbit insertion), the suborbital abort is a secondary abort mode. SPS posigrade range control thrusting is used during this period to ensure a CM landing in the Indian Ocean recovery area. Following SPS range control thrusting, the SM RCS is used to establish the correct CM/SM separation attitude and then to provide the separation velocity. Following separation, the CM RCS is used to orient the CM for a constant bank angle entry by utilizing approximately half-lift capability.

3.2.6.3 Aborts to Orbit

The abort-to-orbit abort mode is designated as the primary abort mode from S-IVB ignition to completion of the earth parking orbit insertion maneuver and is the secondary mode from LES jettison to S-IVB ignition. (The suborbital abort is secondary when SPS range control thrusting is required to obtain a satisfactory landing area.) However, all time-critical aborts during this time period require that the suborbital abort mode be selected.

3.2.6.3.1 Ground Rules

Launch vehicle shutdown must precede initiation of spacecraft abort action. The S-IVB or SM propulsion and guidance systems may be used for the abort-to-orbit mode. For those cases where SPS thrusting is required, the CSM abort separation procedure will utilize both a SM RCS and SPS burn of fixed initiation time, ΔV, and attitude.

Aborts-to-orbit after S-IVB ignition (primary abort mode) will utilize the SPS. All time-critical aborts will be suborbital. Aborts-to-orbit prior to S-IV ignition (secondary abort mode) are dependent on the following time constraints:

a. Aborts initiated prior to T = 140 seconds must utilize the S-IVB propulsion plus the SM propulsion.
b. Aborts initiated after T = 410 seconds but prior to T = 490 seconds must utilize the S-IVB propulsion.
c. Aborts initiated after T = 490 seconds must utilize the SPS.

The abort-to-orbit mode requires that a fixed amount of SM propellant be reserved for de-orbit capability.

3.2.6.3.2 Mode Description

The advisability of aborting to orbit, once the capability exists, depends upon the malfunction. If crew safety is not diminished, one advantage of aborting to orbit is the greater flexibility in selecting a landing site. Figure 3-17 shows the sequence of events during an abort to orbit.

The S-IVB or SPS provides the necessary thrust to place the spacecraft into a 100-n. mi. circular earth orbit; however, the vehicle must be selected as dictated by the time of abort, the capability of each vehicle, and the propellant that must be reserved for de-orbit.

The SM-SPS can supply the necessary ΔV to place the CSM in the parking orbit from an abort occurring about 490 seconds after liftoff. (An SPS abort-to-orbit from an abort occurring at T = 490 seconds is accomplished by burning the SPS for approximately ( ) seconds.) The S-IVB can supply the necessary ΔV to place the CSM in the parking orbit from an abort occurring after about 410 seconds from liftoff. (An S-IVB abort-to-orbit from an abort occurring at T = 410 seconds is accomplished by burning the S-IVB for approximately ( ) seconds.)

Both the S-IVB and SPS must be used for an abort-to-orbit occurring during the period from LES jettison to T = 410 seconds. The abort procedure requires an S-IVB burn followed by an SPS burn.

3.2.6.4 De-orbit Aborts

The only abort possible from S-IVB cutoff at the time of earth parking orbit insertion to the ignition of the SPS for the nominal de-orbit maneuver is an early de-orbit maneuver.

3.2.6.4.1 Ground Rules

During thrusting periods, the initial abort action is to shutdown the SPS or S-IVB as applicable. The spacecraft must consist of only a CSM at the time of initiation of de-orbit thrusting when the SPS is utilized. The de-orbit maneuver will utilize a single propulsive burn at a fixed retrograde attitude and for a specified ΔV. When time permits, the de-orbit maneuver will be initiated at a time to permit entry to one of the planned recovery areas, preferably the primary recovery area. The CM/SM separation procedure will utilize a SM RCS translation and roll maneuver.

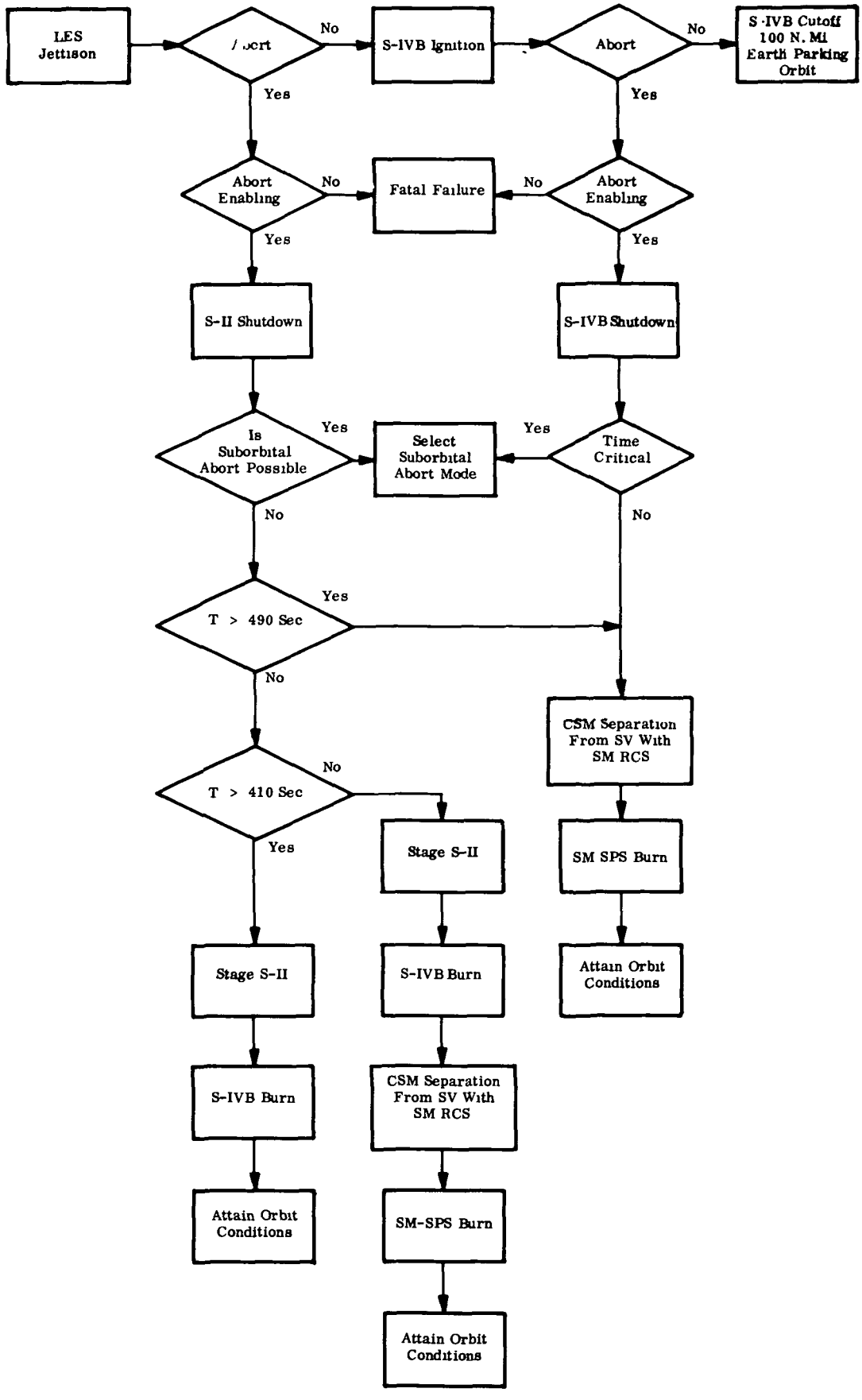

Figure 3-17. Abort to Orbit

For aborts occurring from earth parking orbit insertion to the cutoff of the S-IVB at the completion of the first transfer orbit insertion maneuver:

a. The primary abort mode utilizes the SPS to accomplish the de-orbit maneuver.
b. The secondary abort mode utilizes the SM RCS to accomplish the abort maneuver.

For aborts occurring during the period from S-IVB cutoff (at the completion of the first transfer orbit insertion maneuver) to S-IVB re-ignition (for the second transfer orbit insertion):

a. The primary abort mode utilizes the S-IVB to accomplish the de-orbit maneuver. The S-IVB-IU-SLA-LEM is staged immediately after the de-orbit thrusting.
b. The secondary abort mode utilizes the SPS to accomplish the de-orbit maneuver.

For aborts occurring during the period from S-IVB ignition for the second transfer orbit insertion to ignition of the SPS for the nominal de-orbit maneuver, the only propulsion unit available for the de-orbit maneuver is the SPS.

3.2.6.4.2 Mode Description

A detailed sequence of events during a de-orbit abort is shown in Figure 3-18. If abort action is required during the period between earth parking orbit insertion and S-IVB cutoff at the completion of the first transfer orbit insertion burn, the de-orbit abort procedure requires the shutdown of the S-IVB (if thrusting), separation of the CSM from the launch vehicle, SM RCS orientation of the CSM for the de-orbit maneuver, and de-orbit thrusting by the SPS (primary mode) or SM RCS (secondary mode).

If abort action is required between S-IVB cutoff following the first transfer orbit insertion maneuver, and S-IVB ignition for the second transfer orbit insertion maneuver, the de-orbit abort procedure requires:

a. For the primary abort mode, orientation of the space vehicle for the de-orbit maneuver; de-orbit thrusting by the S-IVB; S-IVB-IU-SLA-LEM staging; and, before entry, CM/SM separation.
b. For the secondary abort mode, staging of the S-IVB-IU-SLA-LEM; SM RCS orientation of the CSM for the de-orbit maneuver; and de-orbit thrusting by the SPS.

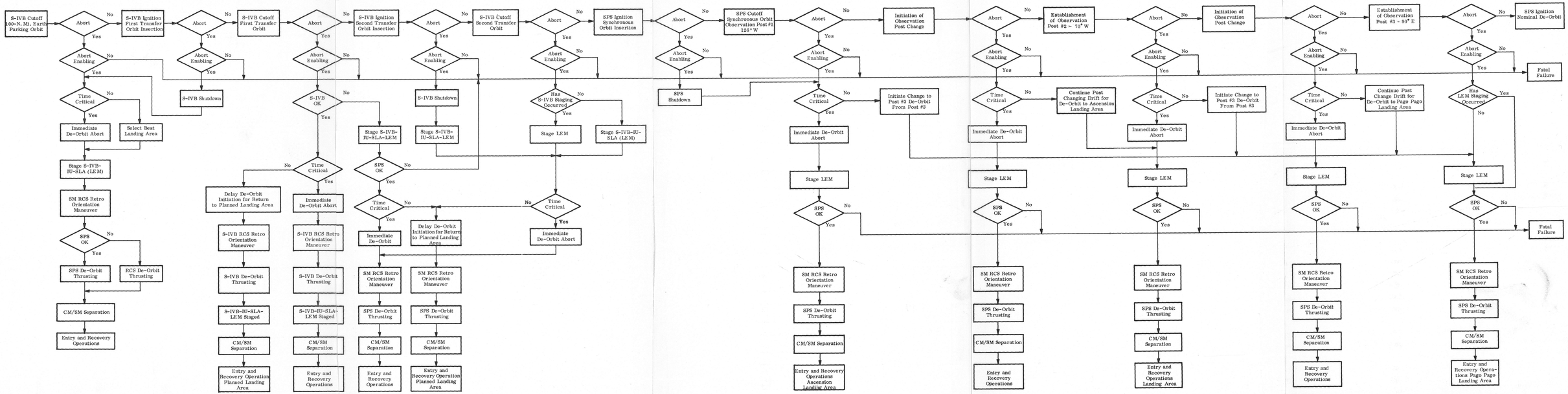

Figure 3-18. De-orbit Aborts

3-77-1 3-77-2 3-77-3 3-77-4

3-78

If abort action is required from S-IVB ignition for the second transfer orbit insertion maneuver to the ignition of the SPS for the nominal de-orbit maneuver, the procedure requires: that the LEM be staged or that the S-TUB-IN-SLA-LEM be staged (if before completion of transposition and docking); that the SPS be shutdown (if the abort occurs during the SPS synchronous orbit insertion maneuver); that the SM RCS be used to orient the CSM for the de-orbit maneuver; and that the SPS be used for de-orbit thrusting.

Time critical aborts require immediate initiation of the de-orbit maneuver with minimum consideration given to the resulting landing area. However, the SPS or SM RCS may be used to alter the earth return trajectory if time and ΔV constraints permit. In non-time critical situations, the de-orbit maneuver can be delayed, if necessary, to return to an adequate landing area.

SECTION 4

MISSION 4

NEAR POLAR RETROGRADE SURVEY

4.1 MISSION DESCRIPTION

4.1.1 MISSION SUMMARY

4.1.1.1 Introduction

The Near Polar Retrograde Survey mission will deploy multispectral sensors on a global basis, collect scientifically and commercially useful data on uncharted regions, and provide further development of advanced meteorological equipment. The mission will utilize a spacecraft, consisting of a modified Apollo Block II CSM, a modified LEM, and a crew of three, placed in a nominal 200-n. mi. circular earth orbit at a 96.5-degree inclination for 45 days (see 19).

4.1.1.2 Configuration

4.1.1.2.1 Launch Vehicle

The launch vehicle used for this mission will be the Saturn V, a multistage vehicle consisting of three propulsive stages (S-IC, S-II, and S-IVB) and a nonpropulsive module, the Instrument Unit (IU) (see 101-6).

4.1.1.2.2 Spacecraft

The spacecraft for this mission will use the Block II CSM Modification D, a modified LEM, LES, and a spacecraft LEM adapter, with added expendables and subsystems to permit operations for 45 days (see 101-2).

4.1.1.3 Experiments Summary

There are 12 types of experiments planned for this flight, utilizing multispectral sensor earth sensing techniques using ground-truth sites, for collecting data of scientific and commercial value from uncharted regions. Experiments in the development of space meteorological equipment are performed, and biomedical and behavioral experiments

are continued. Mapping and radar equipment represent approximately one-half of the total experiment equipment weight, which is about 5,200 pounds (see 35).

The total volume required for experimental equipment is 544 cubic feet, and the average power required is 487 watts with peaks to 1160 watts.

4.1.1.4 Operations

The spacecraft will be launched from Cape Kennedy Launch Complex 39A on a 146-degree launch azimuth into the 200-n. mi. circular, 96.5-degree earth orbit by the Saturn V launch vehicle (19).

The launch into the 200-n. mi. orbit requires that all three stages of the launch vehicle be used. To obtain the desired inclination of 96.5 degrees, yaw steering will be utilized during S-IC and S-II thrusting. Insertion into the 200-n. mi. earth orbit is accomplished by the S-IVB (see 111).

After insertion, the crew performs subsystems tests and conducts IMU alignments and orbital landmark navigational sightings. The orbit ephemeris is determined and evaluated, with coordination and verification by MSFN.

After the orbit ephemeris is verified, transposition and docking are accomplished. After hard dock between the CSM and LEM, the S-IVB, including the IU and SLA, is staged. A second IMU alignment is conducted immediately after S-IVB staging. The interlock and LEM are then pressurized and manned. Upon completion of the required LEM subsystems checks, the mission's prime experiments can be initiated.

The initial experiment preparations are completed and the astronauts begin work/rest cycles. The crewmen perform, adjust, and calibrate any experiments or associated apparatus, perform station-keeping duties, and assume their normal responsibilities during the nominal 45-day mission.

During the final orbits of the nominal mission, the astronauts prepare any experiments and LEM supporting subsystems for continuing operation after the departure of the crew and CSM. After all preparations are completed, the crew members transfer to the CSM and secure the hatches. CSM subsystems are then checked out and prepared for the de-orbiting maneuver. The LEM is staged prior to the final orbit.

During the final orbit, close coordination is maintained with MSFN, and the de-orbit maneuver parameters are determined and programmed into the CSM GNS. The CSM is then properly oriented, the cabin and crew are secured, and the countdown for de-orbiting is initiated. At the termination of the countdown, the CSM GNS automatically commands SM RCS ullage ignition and SPS ignition at the proper times.

The SPS retrothrusting for de-orbit inserts the CSM into a descending transfer orbit with a vacuum perigee altitude of 0 n. mi. Following SPS cutoff, preparations for SM staging and CM re-entry are started. The CSM is properly oriented and the SM is staged - staging occurring 10 minutes prior to re-entry. The CM is properly oriented and final preparations for re-entry are completed immediately after SM staging.

The CM re-enters the atmosphere of the earth at a nominal altitude of 400,000 feet. The CM primary entry guidance system controls the trajectory from 400,000 feet to an altitude of 24,000 feet, where drogue chute deployment occurs. Main chute deployment follows at approximately 11,000 feet, and earth landing occurs shortly thereafter.

4.1.2 MISSION OBJECTIVES

4.1.2.1 Experiment Objectives

The experiment objectives for this mission are to deploy multispectral sensors on a global basis for (1) verification of earth-sensing techniques using ground-truth sites; (2) collection of scientifically and commercially useful data of uncharted regions; and (3) further development of advanced space meteorological equipment (see 35).

The following is a list of experiments (see 33 and 35) to be conducted during Mission 4:

a. 0100 Biomedical (excluding 0101 and 0102).
b. 0200 Behavioral.
c. 0501 Radiation Environment Monitoring.
d. 0802A High-Resolution Infrared Radiometer.
e. 0802B Test of Infrared Scanning Spectrometer.
f. 0802C Test, Calibration, and Evaluation of Microwave Spectrometer.
g. 0802D Test of a Prototype Star-Tracker.
h. 0901 Multispectral Target Characteristics.
i. 0902 Synoptic Earth Mapping.
j. 0903 Multifrequency Radar Imagery.

k. 1405 Large Solar Cell Array.

l. 1407 Onboard Guidance and Navigation.

4.1.2.2 Flight Systems Objectives

4.1.2.2.1 Launch Vehicles

The Saturn V launch vehicle will be utilized (see 9-1 and 9-10) to place the spacecraft payload into the 200-n. mi. mission orbit at a 96.5-degree inclination with zero-degree eccentricity, from an initial launch azimuth of 146 degrees. All three Saturn V stages (S-IC, S-II, and S-IVB) will be used. Yaw steering will be used during S-IC and S-II thrusting.

4.1.2.2.2 Spacecraft

The spacecraft objectives are to carry the mission experiments and the crew of three to the specified mission orbit, return the crew and data samples, and provide operational support for the crew and experiments during the 45-day mission.

4.1.2.2.3 Crew

The crew will consist of three astronauts (see 9-4) who are expected to participate in the performance of experiments beyond merely checking out and operating equipment. They will help to ensure the scientific and technical validity of experiments and provide partial evaluations of results (see 13).

4.1.2.3 Ground Support Objectives

4.1.2.3.1 Mission Command Control

The mission command repsonsibility will remain the same as for the basic Apollo mission, with the ability to pass the mission command to the onboard crew commander for all but the critical mission phases.

4.1.2.3.2 Launch Support and Control Operations

The Launch Control Center and the normal complement of equipment at Pad 39A (see 7) will be utilized for this mission. Modification to launch facilities and checkout operations will be minimized (see 9-13).

4.1.2.3.3 Tracking and Communications Operations

All MSFN remote sites supporting this mission will be remoted directly to MCC-Houston, with sufficient data routed to MCC-Houston in real-time to preclude the necessity for remote site controllers (see 15-11). Tracking and communications are required continuously throughout Saturn V powered flight (see 15-66). This objective will require analysis to determine how it can be best met; the launch azimuth and yaw steering results in a launch trajectory significantly different from those normally employed at Cape Kennedy.

4.1.2.3.4 Recovery Operations

The mission recovery operations will be similar to those of the basic Apollo mission, with the Pacific Recovery Range remaining the primary recovery area (see 24). The potential use of recovery/rescue aircraft will be considered in lieu of continuous assignment of ships for world-wide coverage of abort conditions (see 9-70). The high inclination of the orbit plane requires that planning for the normal and abort condition re-entry sites, be determined on the basis of further trajectory computations.

4.1.2.3.5 Post-flight Operations

The objectives for post-flight operations, involving such activities as data reduction, equipment performance review, crew performance, and ground support performance reviews, have not been considered in the AAM program documentation.

4.1.3 MISSION GROUND RULES

4.1.3.1 Earth Launch Phase

Launch Complex 39A at Merritt Island Launch Areas (MILA) will be used for the launching of Mission 4 (see 1-31).

The launch azimuth for this mission will be 146 degrees (see 49-10). No data is available on range safety launch azimuth limits. The minimum operational earth launch window will be 2-1/2 hours (see 24-4).

Continuous tracking is required from launch to three minutes after insertion into the initial orbit (see 91).

Yaw steering in the S-IC and S-II modules of the Saturn V launch vehicle will be utilized to place the spacecraft into the desired orbit.

4.1.3.2 Mission Orbit Phase

4.1.3.2.1 Orbit Characteristics

The nominal mission orbit will be a near-circular orbit at a 200-n. mi. altitude (see 19). For the purposes of this study the orbit inclination will be 96.5 degrees (see 49-10).

4.1.3.2.2 Ground Network

The flight mission is under control of the flight director, who, under normal circumstances, will delegate control of the scientific program after the spacecraft has achieved its designated flight path (see 14-55). One four-minute contact per orbit with MSFN is required (see 91). The time between two successive contacts will not exceed two hours (see 91).

4.1.3.2.3 Spacecraft Operations

There will be no natural lighting constraints on transposition and docking (see 29-3) except that the LEM docking tunnel must be illuminated. Ground rules on communications requirements during transposition and docking cannot be written at this time. Docking maneuvers will be controlled manually.

The design of the spacecraft will permit extravehicular transfer of the crew between the CM and LEM in either direction, in both the docked and undocked condition (24-3).

The crew will have control of the S-IVB reaction jets for attitude control from earth orbit insertion to separation of the CSM from the S-IVB/LEM (see 24-2). The SPS will be used for de-orbiting (see 14-63).

4.1.3.2.4 Crew Operations

At least one astronaut in the CM will be wearing a spacesuit at all times during flight. The faceplate may be open or closed (see 24-2). Not more than two astronauts will be off duty at any one time during the mission (see 76-5).

4.1.3.2.5 Experiment Scheduling

Ground rules for experiment scheduling cannot be provided at this time. Sample scheduling has been prepared as described in paragraph 4.3.2 which provides the crew task data for experiments, including scheduling, priorities, and time allotments for the various experiment functions.

4.1.3.3 Re-entry Phase

The following are the re-entry trajectory characteristics of Mission 4:

a. Vacuum perigee - 0 n. mi. (see 49-2).
b. Re-entry corridor - not defined at this time (see 29-7).
c. Re-entry range - not defined at this time (see 29-7).
d. Re-entry inclination limits - not defined at this time.
e. Earth landing constraints - none.
f. Cross-range maneuvering capability - not required in the nominal entry trajectory (see 24-5).

The Pacific recovery range is the primary recovery area.

One four-minute preretro MSFN contact is required within 90 minutes prior to retro; one four-minute post-retro contact is required not later than seven minutes after retro (see 91).

4.1.4 WEIGHTS SUMMARY

The spacecraft weights for Mission 4 (518) are summarized in Table 4-1. The table is divided into three sections: Command Module, Service Module, and LEM Laboratory. For each module the major categories are listed and referenced.

The Command Module is an NAA Block II Modification Configuration D (see 45).

The Service Module is an NAA Block II Modification Configuration D (see 45), with the exception of RCS fuel loading, which is based on GAEC AES studies, and the SPS retro-fuel, which is based on computations reported in Appendix B.

The LEM laboratory weights are based on GAEC experiment assignments (see 31), as altered by Bellcomm (see 35) to meet margin requirements.

Table 4-1

Mission No. 4 - Near-Polar Retrograde Survey
Flight 518 - Spacecraft Mission Orbit Injection Weights Summary

Module	Weight (Pounds)		Reference Number and page
1. Command Module			
Empty Weight	9,034		(45) No page
Useful Weight	1,269		(45) No page
Total	[10,303]		
2. Service Module			
Empty Weight	8,921		(45) No page
Reaction Control Propellant[1]	2,457		
Experiments RCS Propellant		(2457)	(See Vol 1 Part II)
Excess RCS Propellant		(0)	
Cryogenic Expendables	653		(45) No page
SPS Propellant[2,3]	41,000		
Total	[53,031]		
3. LEM/LAB			
Structure	2,452		(33) 1.11.24
SCS	43		(33) 1.11.24
NGS	120		(33) 1.11.24
Crew Provision	327		(33) 1.11.24
RCS	886		(33) 1.11.24
(Propellant)		(575)	(33) 1.11.24
Controls and Displays	485		(33) 1.11.24
Instrumentation	1,091		(33) 1.11.25
EPS	4,798		(33) 1.11.25
(Reactant and O_2)		(2438)	(33) 1.11.25
Communications	32		(33) 1.11.25
Experiments	5,214[4]		(35) paragraph 5.5
Spares	96		(33) 1.11.25
Total	[15,544]		
Total Spacecraft Weight[5]	78,878		
Maximum Spacecraft Weight into Orbit	110,000[6]	141836[7]	(See 111)
Margin	31,122	62958	

1. SM RCS propellant tanks capacity.
2. SPS propellant tanks capacity.
3. This figure includes a 10 percent ΔV reserve.
4. The experiment package weight represents a reassignment of Experiment 1407 (weight - 369 pounds) from Flight 218 to Flight 518.
5. Summation of all square bracketed numbers.
6. Stack limit above IU and S-IVB stage (111) No page.
7. No stack limit (111) No page.

4.1.5 EXPERIMENT LOAD DESCRIPTION

4.1.5.1 Loads

The experiment loads for Mission 4 are summarized in Table 4-2.

Twelve types of experiments are planned for the mission. The total weight of the experiment equipment is about 5200 pounds, of which half is for mapping and radar experiments.

4.1.5.2 Support

For experiments with IR scanning and microwave spectrometers, the ground-based operational support consists of consultation with instrumentation personnel to maintain adjustment and calibration of instruments. For experiments with the prototype star-tracker, the spacecraft attitude and trajectory must be calculated accurately.

Ground support requirements for synoptic earth mapping are described in the Mission 1 paragraph 1.1.5. These and other experiments involving observation of the earth surface in various modes will provide data for topographical maps of the desired height resolution, and reconstitution of full-color and modified false-color photographs. For the radar mapping experiments, the coverage must be identical and simultaneous with the other observation methods. During this experiment, the radar altimeter antenna must be oriented in the direction of the earth.

The test and checkout of a prototype onboard navigation system will entail observation of landmarks and stellar objects, orientation of the vehicle, and sightings through the sextant or telescope. Periodic ground station tracking is required for orbital parameter upgrading of the on-board computations. Ground stations will assess the test data. Some degree of orbital maneuvering is planned to evaluate the effect of such motion on navigation accuracy.

Table 4-2

Experiment Load Summary

(35-18) (35-18) (35-18)

Experiment Number	Experiment Category/Abbreviated Title	Total Weight (pounds)	Volume (feet)^3	Reference	Power Watts (average)	Power Watts (peak)
0100	Biomedical - (excluding 0101 and 0102)	301.0	14.5	(33-1-11.5)	100.0	50
0200	Behavioral	20.0		(33-1-11.5)		
0501	Radiation Environment Monitoring	46.5	2.64	(33-1-11.5)	6.3	
0802A	High-Resolution Infrared Radiometer	283.0	51.1	(33-1-11.5)	25.0	
0802B	Test of Infrared Scanning Spectrometer	43.0*	5.64*	(33-1-11.6)	330.0*	
0802C	Test, Calibration and Evaluation of Microwave Spectrometer	129.3	7.1	(33-1-11.5)	90.0	5
0802D	Test of a Prototype Star-Tracker	60.0	3.5	(33-1-11.5)	30.0	
0901	Multispectral Target Characteristics	814.0	64.4	(33-1-11.6)	540.0	210
0902	Synoptic Earth Mapping	1481.0	61.6	(33-1-11.6)	415.0	20
0903	Multifrequency Radar Imagery	1112.0	244.2	(33-1-11.6)	854.0	40
1405	Large Solar Cell Array	597.0	80.1	(33-1-11.7)	120.0	750
1407	Onboard Guidance and Navigation	367.0	9.5	(32-1-7.9)	497.0	50
	Totals	5214.0	544.28		487.0	1160
	S/C Capability	*Included in 0901			(33-1-11.14)	

4.2 OPERATION

4.2.1 MISSION PHASE DESCRIPTION

4.2.1.1 Introduction

This mission requires that a crew of three be launched in a basic Apollo Block II CSM and a modified LEM into a nominal 200-n. mi. circular earth orbit, at a near-polar 96.5-degree inclination. The mission duration is 45 days.

The mission consists of seven major phases for the purposes of analysis and description. Table 4-3 shows each major phase, its initiating event, the initiating event time, the concluding event, and the time duration of each phase. The phase descriptions represent a detailed description of the operations conducted and the crew activities performed during each phase.

It will be noted that many parameters do not currently have assigned values. These parameters were included, based on comparable work in the AMPTF Design Reference Mission, in order to establish the basis for future development of the mission phase descriptions. A blank space, bounded by parentheses where values can be written in, will identify these parameters. Details of the functions during experiment activities are outlined in Appendix D. Scheduling of experiments and flight crew operations are shown in paragraph 4.2.3.

The following is a detailed description* of Mission 4 by phases. Appendix C documents the crew activities prepared from AMPTF-adapted information.

4.2.1.2 Phase 1: Preflight

The preflight phase begins with the securing of the launch umbilical tower (LUT) to the launch pad and concludes with launch vehicle liftoff at T -0.

The LUT is secured to the pad and all interfaces checked, thus all space vehicle-to-ground facilities are verified.

*This description is based on References 24, 26, and 29 and the summary chart of the AAM Earth Orbit Missions, Table 1 of this report.

Table 4-3

Mission Phases, Mission 4, Near-Polar Retrograde Survey

Phase	Initiating Event	Event Time Hr. Min. Sec.	Concluding Event	Duration ΔT Hr. Min. Sec.
1.0 PREFLIGHT	T -485 Hours	T -485 hrs	Liftoff (T -0)	485 hrs
2.0 EARTH ASCENT	Liftoff (T -0)	0.0	Parking Orbit Insertion - S-IVB Cutoff	973.23 sec
2.1 S-IC Thrusting (Yaw Steering)	Liftoff (T -0)	0.0	S-IC Staging	154.37 sec
2.1.1 Coast		150.57 sec		3.8 sec
2.1.2 S-IC Staging		154.37 sec(1)		
2.2 S-II Thrusting (Yaw Steering)	S-IC Staging	154.37 sec(1)	S-II Staging	376.62 sec
2.2.1 LES Jettison		185.57 sec(1)		
2.2.2 Coast		526.19 sec		4.8 sec
2.2.3 S-II Staging		530.99 sec(1)		
2.3 S-IVB Thrusting	S-II Staging	530.99 sec(1)	EOI - S-IVB Cutoff	
3.0 200-N. MI. EARTH ORBIT COAST	EOI - S-IVB Cutoff	16 m 13 sec(1)	Beginning of De-orbit Preparations	45 days
3.1 Initial Coast	EOI - S-IVB Cutoff	16 m 13 sec	CSM/LEM Separation	
3.2 Transposition and Docking	CSM/LEM Separation		Hard Dock	40 min(3)
3.3 Survey Operations	Hard Dock		Beginning of De-orbit Preparations	45 days
4.0 DE-ORBIT	Beginning of De-orbit Preparations	45 days	Re-entry	
4.1 LEM Staging	Beginning of De-orbit Preparations	45 days	SPS Retro Ignition	1 h 30 m(2)
4.2 SPS Retro Thrusting	SPS Retro Ignition	45 d, 1 h, 30 m	SPS Retro Cutoff	12.5 sec
4.3 De-orbit Coast	SPS Retro Cutoff		SM Staging Initiation	17 min
4.4 SM Staging	SM Staging Initiation	45 d, 1 h, 47 m	Re-entry	10 min
5.0 RE-ENTRY	(400,000 Feet) Re-entry	45 d, 1 h, 57 m	Drogue Chute Deployment	10.5 m(3)
6.0 PARACHUTE DEPLOYMENT	Drogue Chute Deployment	45 d, 2 h, 7 m	Touchdown	7 min(3)
7.0 RECOVERY AND RETRIEVAL	Touchdown	45 d, 2 h, 14 m	Retrieval	

1. Reference 49, Table 9.
2. Reference 26, Adapted.
3. Reference 24.

The space vehicle checkout begins with a checkout of the GNS. Included in this subphase are launch vehicle propellant tanking tests, verification of all interface connections, spacecraft fueling operations, and a simulated spacecraft flight. All spacecraft systems are then shut down at T -120 hours.

The precountdown begins 485 hours before the scheduled launch time and continues until terminal countdown, 10 hours prior to launch. During this period, the LEM and CSM receive cryogenic servicing, the GNS of both the CSM and LEM are checked out and activated, and the launch vehicle is loaded with propellant. Other operations include the installation of cabin, crew, and water supplies, and the final installation of LEM scientific equipment.

Ten hours prior to launch, the terminal countdown begins. The CSM is cleaned, sterilized, and closed out. The launch vehicle obtains its cryogenic load while tracking and communications networks are tested. At T -6 seconds, the S-IC engine ignition sequence is started, and at T -0 the vehicle lifts off, ending the preflight phase.

4.2.1.3 Phase 2: Earth Ascent

The earth ascent phase (see Appendix C) begins with Saturn V liftoff, and concludes with the S-IVB cutoff after earth parking orbit insertion.

After a 6-second thrust buildup period from S-IC ignition, the vehicle is released and lifts off (T -0 seconds). The vehicle rises vertically (pitch = 90 degrees) for a 12-second period and then begins a programmed pitch maneuver that smoothly reduces the pitch angle to 52.65 degrees at the burnout point. At 30,000 feet, the dynamic pressure reaches a peak of 690 psf. Yaw steering is used during a portion of the S-IC thrusting period. At S-IC staging at T +154.37 seconds, the vehicle velocity and approximate altitude are 6121 fps and 265,964 feet, respectively. The flight azimuth at S-IC cutoff is 164.6 degrees. The S-IC booster is staged prior to S-II ignition.

During ascent, the commander monitors acceleration, altitude, altitude rates, abort lights, and pitch and roll rates. He also monitors the programmed yaw maneuver on the flight director attitude indicator. The navigator is responsible for monitoring the S-IC ignition and the caution and warning indicators. The navigator also monitors the environmental control system panel. For the duration of the ascent phase, the engineer maintains radio communication with the control center and monitors the critical

temperature flow and events display. In addition, the commander reports the S-IC engine cutoff and staging sequence, using the two-way voice communication link.

After S-IC cutoff the vehicle coasts 3.8 seconds before S-II ignition at T +154.37 seconds. Thirty-one seconds after verifying S-II ignition, the commander confirms the LES jettison. Yaw steering is used for about the first 20 seconds of S-II thrusting period to change the flight azimuth from ( ) to ( ). The flight azimuth at S-II cutoff is -186.5 degrees.

The S-II engines shutdown approximately nine minutes after launch on a signal from the propellant level sensors. The commander monitors this cutoff and staging sequence.

S-II burnout occurs at T +526.19 seconds, with a vehicle velocity and altitude of 14,862 fps and 200 n. mi. feet. At this point, the S-II and the CSM/LEM-S-IVB configuration is allowed to coast for 4.8 seconds before the S-II is staged and S-IVB ignition occurs.

The S-IVB is ignited at T +530.99 seconds. The vehicle performs a programmed pitch maneuver, which terminates with vehicle insertion into a 200-n. mi. earth orbit and cutoff of the S-IVB. S-IVB cutoff occurs at T +973 seconds; vehicle velocity at S-IVB cutoff is 25,228 fps at longitude -88.8 degrees and latitude -3.19 degrees.

4.2.1.4 Phase 3: Earth Orbit Coast

The earth orbit coast phase (see Appendix C) begins with the spacecraft insertion into the 200-n. mi. earth orbit. The phase ends with the initiation of de-orbit preparations.

After S-IVB engine cutoff, the commander and navigator confirm orbit achievement. All three crewmen then adjust their couch restraints and set the controls and displays for earth orbit operations. Using the inflight test system, the navigator and engineer perform a checkout of the spacecraft systems, including guidance and control, environmental control, reaction control, and electrical power. The engineer also continues to maintain communications with Mission Control Center.

The commander stabilizes the spacecraft attitude so that the navigator can perform the IMU fine alignment and landmark navigational sightings. The S-IVB reaction control system is used to orient the space vehicle for landmark acquistion.

Once he has determined and verified the orbit ephemeris, the commander makes the decision for mission continuation in the normal mode.

The spacecraft is then oriented for transposition and docking. This transposition maneuver is initiated by the commander.

When the S-IVB-LEM/CSM separation is assured, the SLA is deployed to aid in the stabilization of the S-IVB-LEM and to expose the LEM. The CSM then translates 35 feet forward, performs a 180-degree pitch maneuver, and translates back to the S-IVB-LEM. The CSM performs docking maneuvers to align the CM-LEM interface and to engage the LEM drogue (X-axis docking tunnel). A maximum of 15 minutes is required from separation to the accomplishment of a soft dock, which occurs at a mission time of 1 hour 30 minutes. After soft docking the navigator secures the docking latches and electrical connections between the two modules, and pressurized the LEM.

Not more than 25 minutes is required from soft dock to the securing of the CM hatches after performing the hard dock.

Upon completion of hard docking, the commander initiates the S-IVB separation from the CSM/LEM configuration, a sequence visually monitored by the engineer. The commander then translates the CSM/LEM from the jettisoned S-IVB stage. The engineer reports the maneuver completion to Mission Control Center.

A maximum of five minutes is required from hard docking to the completion of S-IVB/CSM-LEM separation. S-IVB staging occurs at a mission time of 2 hours 0 minutes at 140 degrees longitude and -60 degrees latitude.

While the engineer continues to record PCM telemetry and maintain ground communications, the navigator proceeds with an IMU fine alignment and the landmark navigation sightings. During these operations, the commander maneuvers the vehicle attitude for landmark acquisition. The navigator next compares his orbit information with ground-computed parameters to ensure accuracy of the onboard calculations.

The engineer prepares to transfer to the LEM for earth orbit operations. The commander checks the LEM pressurization from the CM. The engineer then equalizes the CM and interlock pressure, removes and stows the CM pressure and thermal hatches, and disconnects and stows the probe and drogue. After activating and checking out his

portable life support system, the engineer transfers from the CM to the LEM through the tunnel. Upon completion of an inspection of the LEM cabin, he closes the hatch and initiates communications with the MSFN. After a brief checkout period, experiments which require crew activation or participation are set up.

During each day of the orbit phase, the crew's primary activities include scientific experiments, systems status checks, and navigational sightings. Personal functions are performed in the time remaining.

At the end of the 45-day orbit phase of the mission, the engineer deactivates the LEM systems and transfers to the CSM. At this time, the crew transfers to the LEM all CM and experimental equipment to be jettisoned. Upon completion of these activities, each crew member enters his couch, fastens his restraints, and begins preparation for the LEM jettison.

4.2.1.5 Phase 4: De-orbit

The de-orbit phase (see Appendix C) begins when the crew starts de-orbit preparations, and terminates at CM entry into the earth atmosphere at 400,000 feet.

After orienting the spacecraft for LEM jettison, the commander initiates LEM/CSM separation by igniting shaped charges to sever the LEM/CSM interface. He then translates the CM from the LEM, using the SM reaction control system. The commander maneuvers the spacecraft to the proper attitude while the navigator performs a navigation fix. After completing the IMU fine alignment and a landmark navigational sighting, the navigator computes the onboard trajectory. MSFN supplies orbital parameters for comparison with the navigator's trajectory calculations. The crew then checks all CM systems in preparation for the SPS burn into the de-orbit coast phase.

The LEM is staged at a mission time of T +45 days, 0 hours, and 30 minutes at a geocentric location of 200 n. mi. altitude, -110 degrees longitude, and 60 degrees latitude.

An SPS retrofiring of 12.5 seconds over the arctic inserts the CSM into a descending transfer orbit having a vacuum perigee of zero n. mi. The SPS thrust initiation begins at a mission time of T +45 days, 1 hour, and 30 minutes and at 35 degrees longitude and 0 degrees latitude.

Following verification of the navigator's de-orbit parameters by MSFN, the three crewmen don their pressure suits and adjust their seat restraints. The CSM GNS is programmed, and the CSM is oriented before the crew initiates the countdown for SPS retro-ignition. At the termination of the countdown, the CSM GNS automatically commands SM RCS ullage ignition and SPS ignition at the proper times.

The commander checks the insertion parameters and the CSM systems status, and communicates his findings to MSFN. Next, he prepares for SM jettison by orienting the CSM to the proper attitude and activating the CM/SM separation sequence. During this period, the engineer switches all systems from SM to CM and retracts the antenna and optics. After a final IMU alignment, the CSM is properly oriented and the SM is staged. SM staging occurs 10 minutes prior to re-entry. At the time of SM staging, the spacecraft is 190 miles above the earth's surface, at 15 degrees longitude and 68 degrees latitude.

All three crewmen monitor the CM/SM separation. After SM separation, the navigator and engineer perform a post-separation status check of the CM reaction control system. The commander then maneuvers the CM to the MSFN-confirmed entry attitude. Following a check by the commander of the entry monitor flight indicator, the crew members adjust restraints, check suit umbilicals, and close face plates.

Following SM jettison, a final pre-entry check is concluded, which includes an RCS and GNS entry check, entry monitor systems check, a crew equipment systems check, and a CM orientation for re-entry. Re-entry occurs at a geocentric longitude and latitude of -139 degrees and 65 degrees, respectively.

4.2.1.6 Phase 5: Re-entry

The re-entry phase begins when the CM enters the earth atmosphere, nominally at an altitude of 400,000 feet, and ends at the time of drogue chute deployment, which occurs at an altitude of 24,000 feet.

The re-entry phase begins at a mission time of T +45 days, 1 hour, and 57 minutes at a nominal entry altitude of 400,000 feet, and at a geocentric longitude and latitude of -139 degrees and 65 degrees, respectively. For the preselected landing site target of -157 degrees and 8 degrees, the range to go at the time of entry is ( ) n. mi. The entry velocity, flight path angle, inclination and azimuth are 25831 fps, -1.52 degrees, ( ) degrees, and 196.3 degrees, respectively.

The initial part of the re-entry phase is a full-lift flight from 400,000 feet to the point where the rate of descent is reduced to 700 fps. At this point, trajectory range control is initiated. During trajectory range control, the steering commands are based on a reference trajectory selected to match the guided trajectory predicted range (from the control initiation point to touchdown) to the range requirements to reach the target. The final part of the entry phase consists of an equilibrium glide to an altitude of 24,000 feet, the altitude at which drogue chute deployment occurs.

During re-entry, the commander continues to monitor the CM entry attitude and glide maneuver. The navigator starts the elapse event time clock on a 0.05-g indication. The engineer maintains communications with MSFN recovery forces throughout this re-entry phase. At 100,000 feet altitude, the commander arms the earth landing system and monitors the descent to 25,000 feet. The forward heat shield is jettisoned at an altitude of 25,000 feet, 3 seconds prior to drogue chute deployment, which marks the beginning of the parachute descent phase.

4.2.1.6 Phase 6: Parachute Descent

The parachute descent phase begins at an altitude of 24,000 feet, at the time of drogue chute deployment, and ends approximately 7 minutes later with CM earth landing.

The parachute descent phase begins at a mission time of T +45 days, 2 hours, and 7 minutes, when the CM is at an altitude of 24,000 feet and approximately over the earth landing site location. The CM is at a geocentric longitude of -157° and a latitude of 8° at the time of drogue chute deployment. The CM relative velocity and flight path angle at this point is ( ) fps and ( ) degrees, respectively.

The navigator monitors the forward-heat-shield deployment at 25,000 feet, and the drogue chute deployment 1.5 seconds later. If parachute deployment does not occur automatically, he deploys the chute manually. The drogue chutes are jettisoned and main chutes are deployed at an altitude of 11,000 feet, 58 seconds after the beginning of the phase.

After the navigator activates the VHF system, all three crewmen shut off all power switches not required for landing. The remaining propellant in the CM RCS is then dumped by the navigator. The commander continuously monitors the altitude and descent velocity during this phase, communicating the status of the descent to the MSFN.

Earth landing occurs at 480 seconds after the deployment of drogue chutes. At CM landing the relative velocity and flight path angle are 25 fps and -90 degrees. Respectively. Earth landing occurs at a longitude of -157° and a latitude of 8°.

Recovery communications (VHF) are activated and a final systems check is performed during parachute descent, starting 4 minutes and 49 seconds prior to touchdown.

4.2.1.7 Phase 7: Recover and Retrieval

This phase begins with CM touchdown, and continues until both the CM and crew are onboard a recovery ship. It begins at a mission time of T +45 days, 2-hours, and 14 minutes.

After touchdown, each crew member releases his seat restraints and assesses his physical condition and the structural condition of the CM. While the commander activates the post-landing recovery aids, the engineer checks out the EPS and ECS for post-landing operations. After obtaining survival equipment from the storage area, the crew egresses to await recovery.

4.2.2 CONFIGURATION

4.2.2.1 Introduction

The Mission 4 spacecraft is comprised of the NAA Block II Configuration D Command and Service Modules, and a LEM/LAB, which includes ascent and descent stages.

The configuration of the LEM/LAB and the location of experiment equipment is shown in Figure 4-1. Conceptual sketches of equipment are available for seven types of experiments conducted during this mission, and these are presented in Figure 4-2 (see 11, 40, and 52).

The navigation and guidance experiment (1407) requires periodic spacecraft orientation towards a landmark, a stellar inertial reference, and to local vertical. The LEM optical telescope installation in the upper forward cabin provides conical (60-degree) coverage for the 1407 experiment. The star tracker (1407) installed in the upper forward position of the ascent stage also provides a conical (45-degree) coverage.

Nominal spacecraft attitude is earth-oriented with the spacecraft longitudinal axis located in the orbital plane. This attitude is held within ±5 degrees, except during

experiment periods having specific pointing and stability requirements. Pointing and stability requirements up to 0.03 degree per second can be accommodated by the CSM within the RCS propellant capacity. Additional support is obtained from the LEM RCS.

A viewfinder installation provides hemispherical coverage for all experiments. Structural modifications to the ascent stage are required. The optical sensor is a gimbal-mounted installation utilizing optics located in the descent stage. A 250-foot^2 solar panel assembly, used to supplement laboratory power demands, is stored in the descent stage and deployed by the astronauts during early orbit operations. Cryogenic cooling is supplied by a Block II liquid hydrogen tank.

Adequate internal volume is available to accommodate primary and secondary experiments. All primary experiment displays and controls are located in the forward cabin or are accessible from forward cabin crew positions. A seven-day capacity of lithium hydroxide canisters is stored internally, mounted on the aft bulkhead of the laboratory, and replaced during programmed EVA's.

Supporting electronics and spares are carried in the center bay of the descent stage affording ready access for maintenance.

4.2.2.2 Configuration History

The configuration history for this mission is shown in Figure 4-3. This figure divides the mission profile into three major operation areas. The launch vehicle and spacecraft are depicted as appropriate for each of these major areas. Mission times from the trajectory analysis and mission phasing charts are shown relating time to the various configuration changes.

4.2.2.3 Expendables Weights

4.2.2.3.1 Life Support System

The expendables weights for Mission 4 are summarized in Figure 4-4.

The following weights are ascribed to expendables of the life support system (see 126 and 127).

a. Food - 228 pounds of food are stored on the LEM (see 33-1-11.22). This amount is sufficient to sustain the crew for the mission at a consumption rate of 1.4 pounds per man per day.

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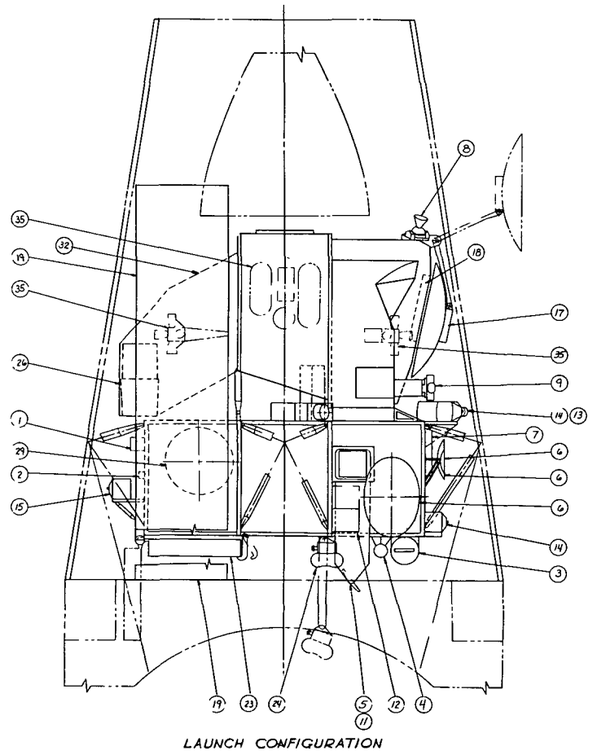
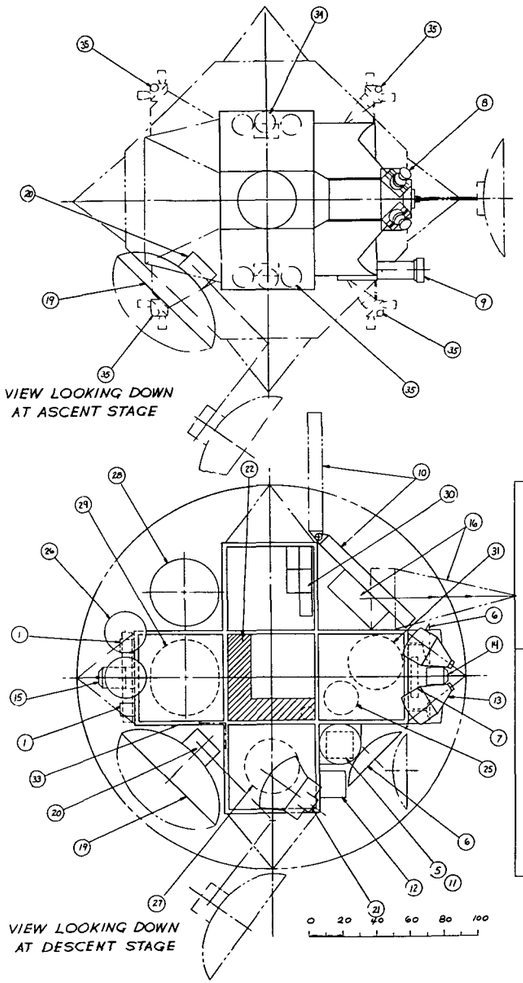
1-11.25

KEY

No.	Experiment	Name
1	501	Electron Spectrometer (2)
2	501	Proton Alpha Spectrometer
3	802A	IR Radiometer (Also use on 901-2B in place of Spectro/photometer)
4	802	Ceramic Sphere
5	901,802B	Use 901 Grating Spectrometer & Telescope (replaces 802B IR, Near IR, and UV-vis spectrometers
6	802C	12-in., 24-in., & 48-in. Antennas
7	802C	Receivers 10-18 GC 40-50 GC 18-26 GC 49-59 GC 26-40 GC
8	802D	Theodolites
9	902,802D	Viewfinder
10	901	Microwave Imager Antenna
11	901,802B	Grating Spectrometer & Dewar (Use on 802B)
12	901	Multispectral Camera
13	902	24-in. Fl Camera (2)
14	902,802A,802B	6-in. Fl Camera (3)
15	902	70-mm Star Camera
16	902	VHF Pulse Radar Antenna
17	902	Microwave Radiometer
18	903	Altimeter
19	903	Antenna (Folded on 8 D FLT only, otherwise 1 piece construction)
20	903	Electronics (4 cu ft)
21	903	Panoramic Camera
22	802,901 902,903	Electronics Equipt Bay
23	1405	Solar Cell Array
24	1405	16mm Movie Camera & TV Camera
25		GOX
26		FCA
27		H_2 (Block II Tank)
28		H_2 (Block III Tank)
29		O_2 (Block III Tank)
30		LiOH
31	901,902 903,802	Cryogenic Cooling
32		Radiators (FCA)
33		Radiators Equipt
34		RCS He (on 11 D only)
35		RCS (on 11 D only)

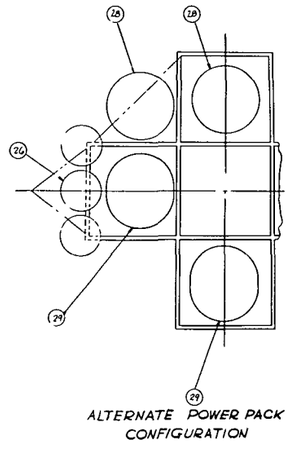


LAUNCH CONFIGURATION

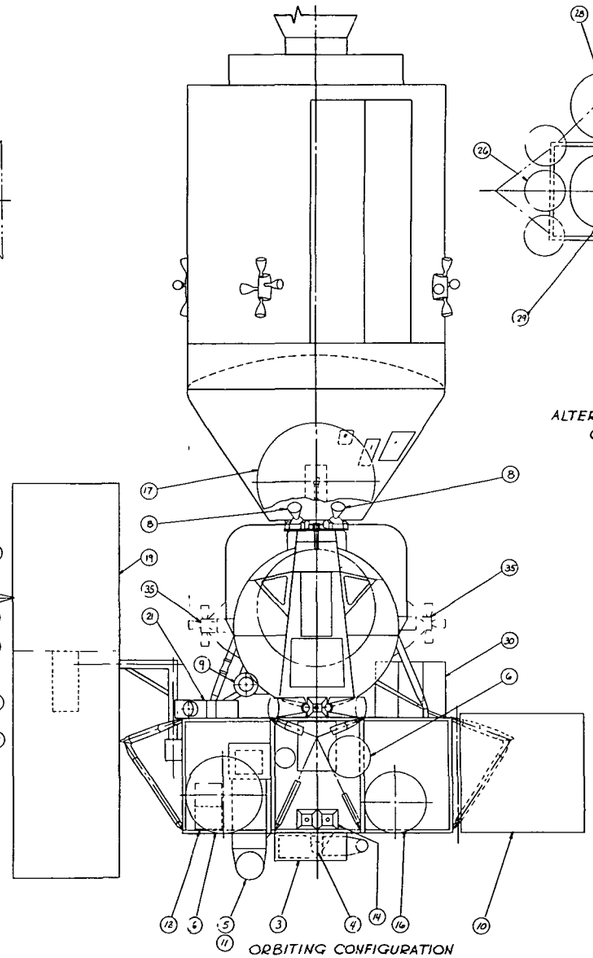


VIEW LOOKING DOWN AT ASCENT STAGE

VIEW LOOKING DOWN AT DESCENT STAGE



ALTERNATE POWER PACK CONFIGURATION



ORBITING CONFIGURATION

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Figure 4-1. Spacecraft Configuration and Experiment Location (Sheet 1 of 2) (See 33 1.11.25)

4-21-1

2

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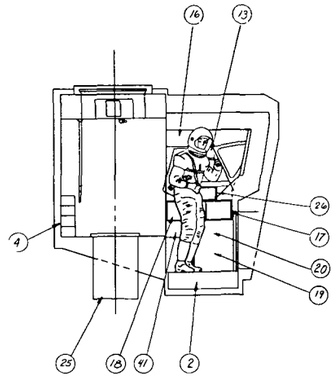
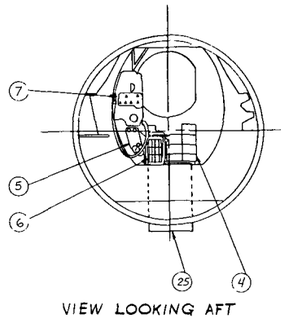
KEY

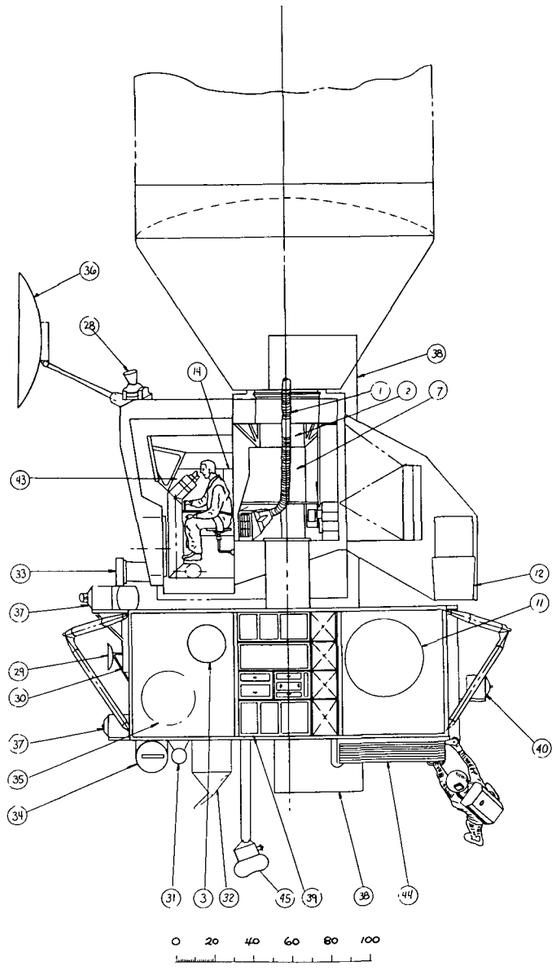
No.	Experiment	Name
1		Air Duct
2		Food
3		GOX Tank (Accumulator)
4		LiOH
5		Water Module
6		Fan
7		ECS
8		Seat
9		Flood Lights
10		Adjustable Work Top
11		O_2 (SOX)
12		Fuel Cell Assy
13		Pwr Gen & Dist Panel
14		Cryogenic Panel
15		Computer Input Panel
16		Circuit Breaker Panel
17		Digital Computer
18		Data Receiver
19		Film & Tape Storage
20		Log & Data Book
21		Tape Recorder
22	0100&0200	Biomed Incubator
23		Biomed Refrigerator
24		Biomed Panel
25	0100&0200	Biomed Storage
26	0501	Pulse Height Analyzer
27	0501	Radiation Envir Monitor Panel
28	0802D	Theodilite
29	0802C	Antenna
30	0802C	Receivers
31	0802	Ceramic Sphere
32	901,802	Telescope
33	902,802D	Viewfinder
34	901-2B, 802A	IR Radiometer
35	901,802	Cryogenic Tank (Cooling)
36	902	Microwave Radiometer
37	902	Mapping Cameras
38	903	Antenna
39	901,902, 903	Electronics
40	902	70mm Star Camera
41	1405	Pyrheliometer
42		Multimeter
43		Solar Cell Array Readout Panel
44		Solar Cell Array
45	1405	Camera



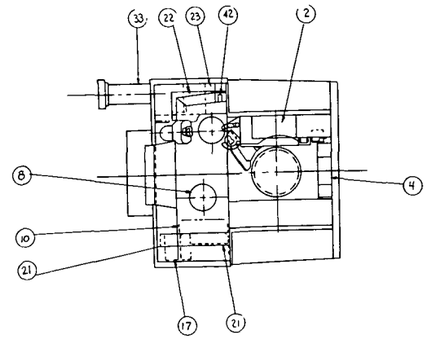
VIEW LOOKING OUTBOARD
LEFT HAND SIDE



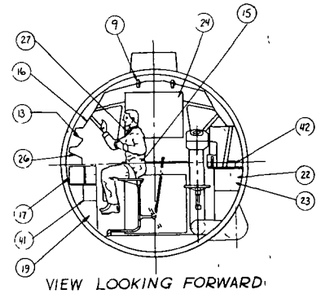
VIEW LOOKING AFT



VIEW LOOKING OUTBOARD
RIGHT HAND SIDE



PLAN VIEW



VIEW LOOKING FORWARD

Figure 4-1. Spacecraft Configuration and Experiment Location (Sheet 2 of 2)
(See 33 1.11.26)

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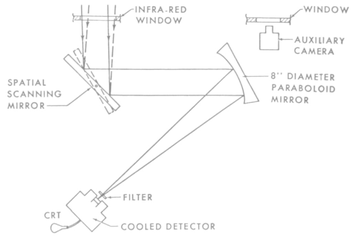
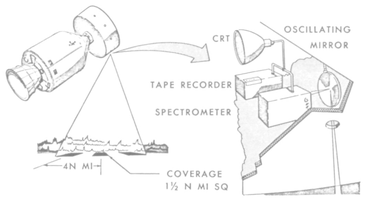
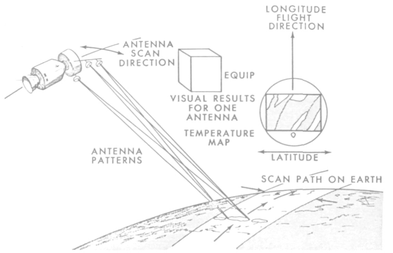
HIGH RESOLUTION INFRA-RED RADIOMETER
0802A



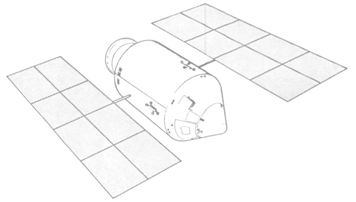
IR SCANNING SPECTROMETER - 0802B



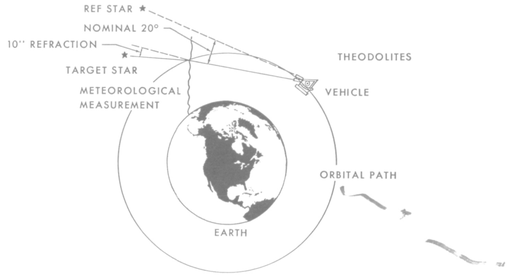
MICROWAVE SPECTROMETER - 0802C



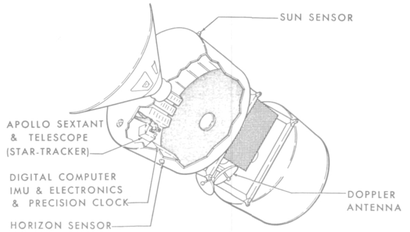
LARGE SOLAR CELL ARRAY - 1405



PROTOTYPE STAR TRACKER - 0802D



ON-BOARD NAVIGATION & GUIDANCE - 1407



MULTI-FREQUENCY RADAR IMAGERY - 0903

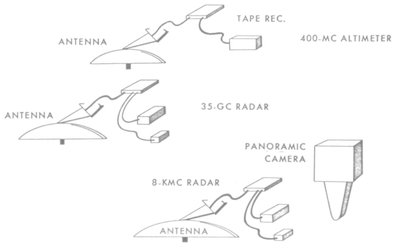


Figure 4-2. Conceptual Sketches of Experiment Equipments (See 11, 13)

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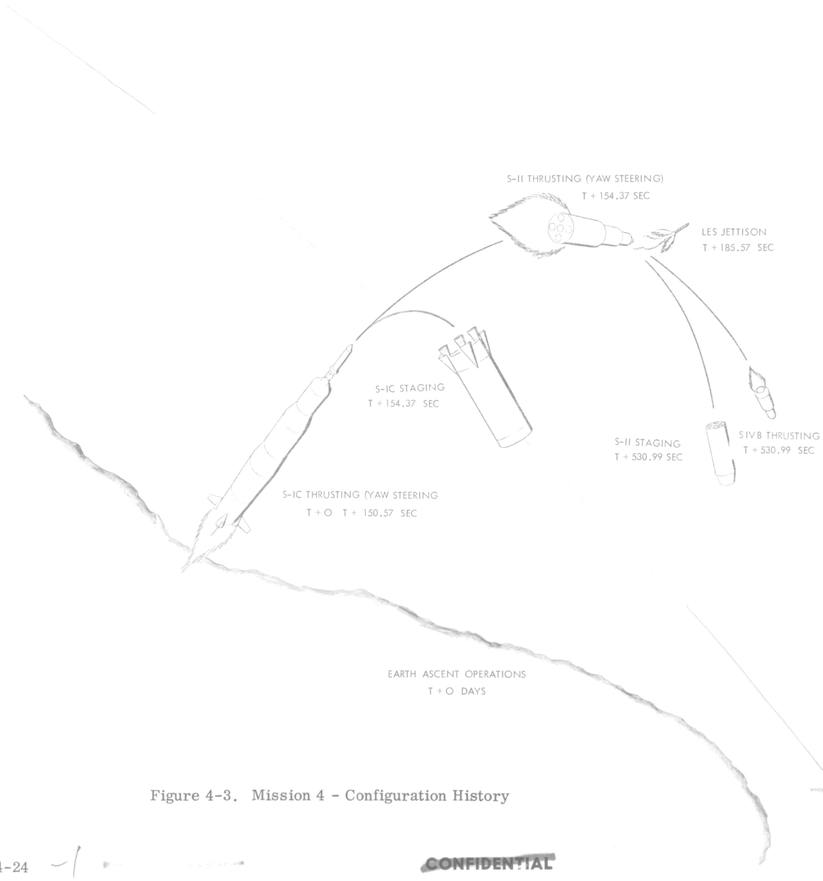

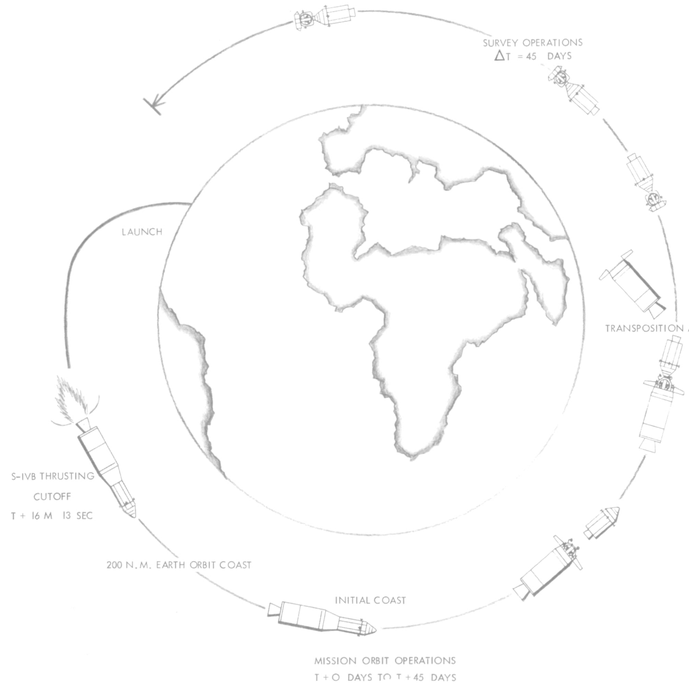

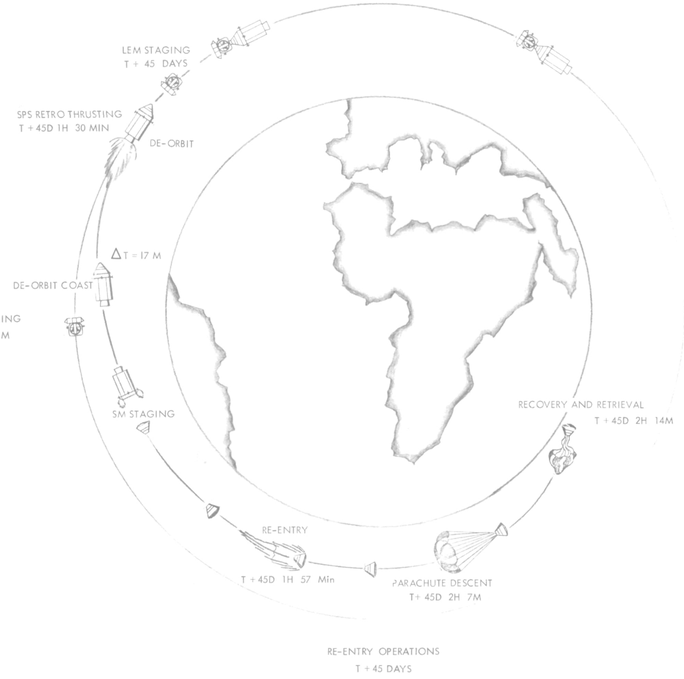

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Figure 4-3. Mission 4 - Configuration History

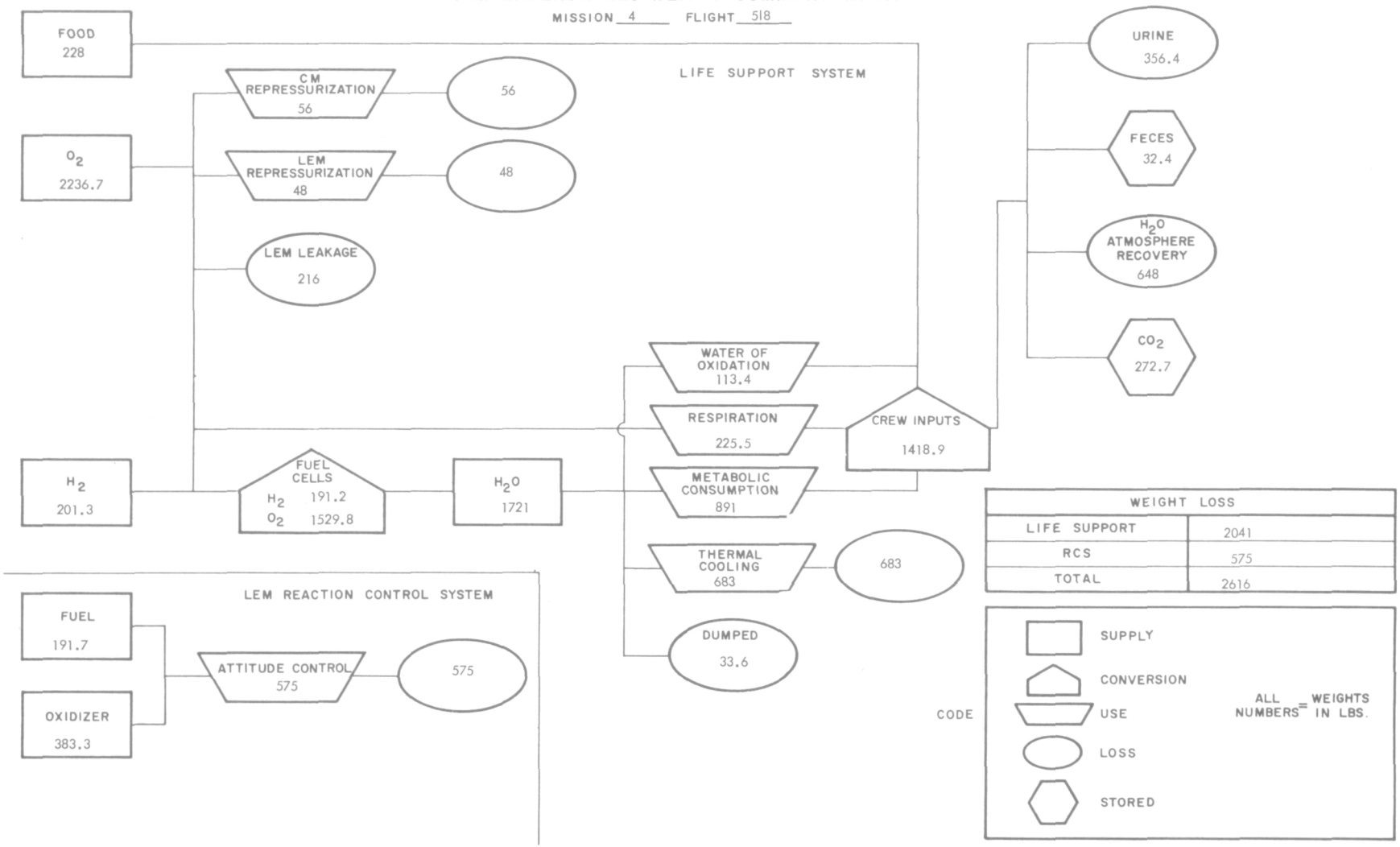

Figure 4-4. Expendables Weights Summary Chart (Sheet 1 of 2)

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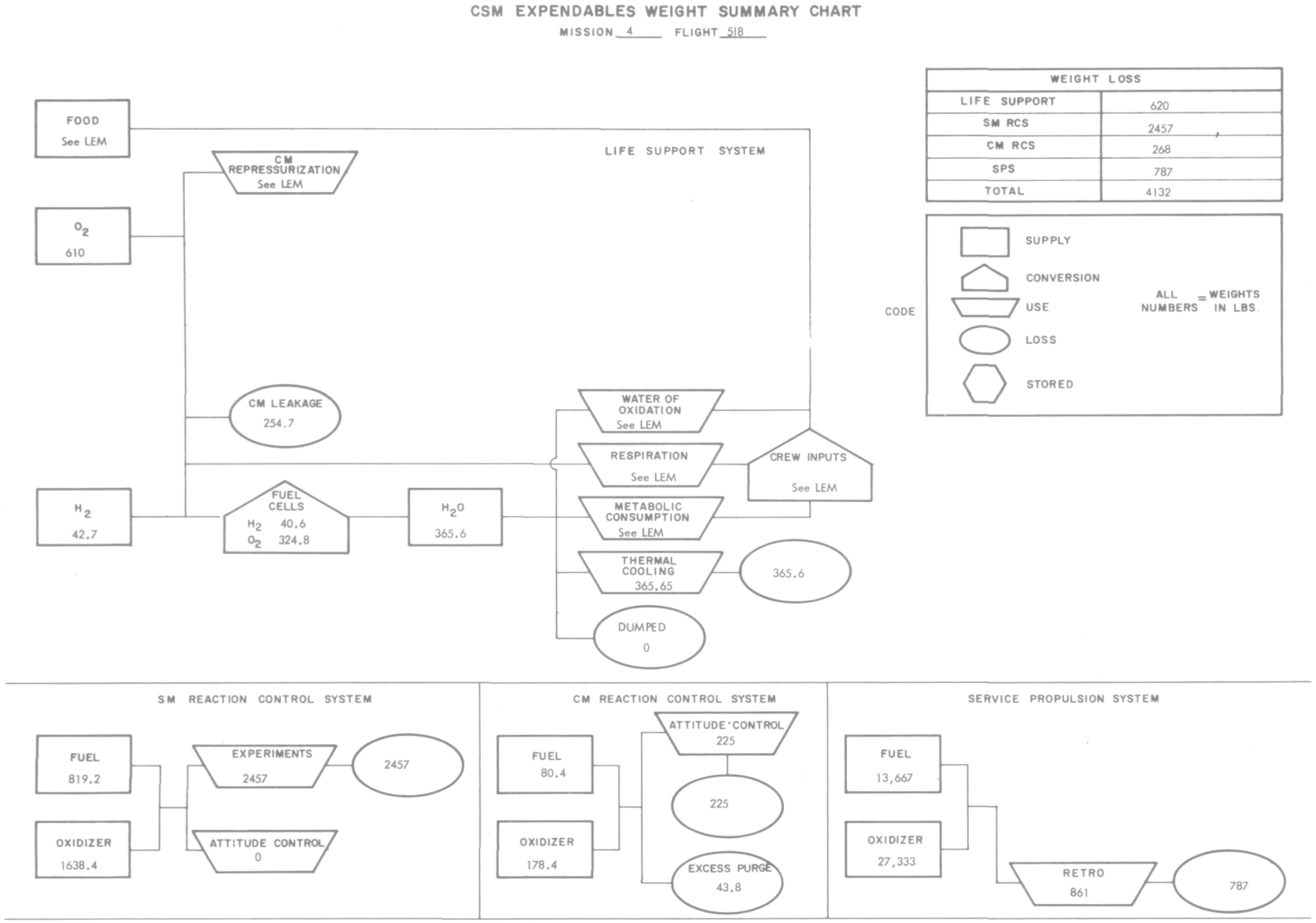

Figure 4-4. Expendables Weights Summary Chart (Sheet 2 of 2)

b. Oxygen - The two CSM oxygen tanks are filled to 652.8 pounds (see 80-167). The two LEM Apollo X oxygen tanks (see 33-1-11.14) are loaded with 2137 pounds.

c. Hydrogen - The two CSM hydrogen tanks are filled to 58.24 pounds (see 80-167). The two LEM Apollo X hydrogen tanks (see 33-1-11.14) are loaded to 201 pounds.

d. Repressurization - For extravehicular activity, the CM atmosphere is evacuated overboard. It was assumed that the CM and LEM have no airlocks and that the LEM atmosphere also is dumped. LEM repressurization requires six pounds of oxygen; the CM requires seven pounds (see NAA Training Manual). Five repressurizations are required during the mission (see 33-1-11.14).

e. Leakage - The oxygen leakage rate for the CM is 5.66 pounds per day, for the LEM 4.8 pounds per day. The total oxygen leakage is 216 pounds for the LEM and 255 pounds for the CM, based on a cabin pressure of 5 psia.

f. Respiration - The oxygen required for crew respiration is 1.67 pounds per man per day, which results in a requirement of 225.5 pounds to be supplied by the LEM.

g. Fuel Cells - The CSM and the LEM each have three fuel cell assemblies (see 80 and 33-1-11.14). The hydrogen and oxygen supply for the CSM cells is the original supply less amounts required for respiration, repressurization, and leakage replacement.

h. Water - The fuel cell production of water varies, with power requirements, from 28.8 to 82.8 pounds per day for each of the three assemblies. Hydrogen and oxygen released to the fuel cells combine to form water.

i. Water of Oxidation - The water needed to reconstitute powdered food is 0.035 pounds per man per hour, for a total of 113.4 pounds.

j. Metabolic Consumption - Drinking water is consumed at a rate of 0.275 pounds per man per hour, for a total of 891 pounds.

k. Thermal Cooling - Water required for LEM cooling is 683 pounds (see 33-1-11.14). An equal amount is assumed for the CM.

l. Water Dumped - Excess water produced by the fuel cells is vented overboard.

m. Crew Inputs - The total weight of crew inputs is as follows:

(1) Urine - 0.11 pounds per man per hour, totaling 356 pounds, which is vented overboard.

(2) Feces - 0.24 pounds per man per day for a total of 32 pounds, which is stored aboard.

(3) Water Vapor (expiration and perspiration) - 0.2 pounds per man per hour, a total of 648 pounds, which is recovered and dumped.

(4) Carbon Dioxide (human respiration) - 2.02 pounds per man per day (see SID 63-1370-2). The total, 273 pounds, is stored onboard and absorbed by lithium hydroxide filters.

4.2.2.3.2 SM RCS

The two LEM SM RCS fuel tanks per quad are loaded to 819 pounds. The two LEM SM RCS oxidizer tanks per quad are loaded to 1638 pounds.

The RCS requirements for attitude control during the experiments partially fulfilled is 2457 pounds of fuel-oxidizer mixture of a ratio of one to two. No reactants are available for normal attitude control functions.

a. CSM booster separation.
b. Ullage maneuver prior to SPS ignition.
c. Separation of the SM and CM.

4.2.2.3.3 CM RCS

The two CM RCS fuel tanks (see 80-85) are filled to 90.4 pounds. The two CM RCS oxidizer tanks are filled to 178.4 pounds.

Of the total, 225 pounds of reactants is required to provide:

a. Maneuver capabilities during entry after SM separation.
b. Maneuver capabilities in a high-altitude abort after LES jettison.
c. Three-axis control to stabilize CM motion prior to deployment of ELS parachutes.

Fuel and oxidizer left in the tanks after parachute deployment and final attitude correction is dumped before touchdown.

4.2.2.3.4 SPS

The single SPS fuel tank (see 80-viii) has a capacity of 15,435 pounds; the load for this flight is 13,667 pounds. The single SPS oxidizer tank has a 31,620 pound capacity; the load for this flight is 27,333 pounds. A reactant weight of 787 pounds is used by the SPS for the retro maneuver (calculated).

4.2.2.3.5 LEM RCS

The two LEM RCS fuel tanks (see 33-1-11.14) have a 204.8 pound capacity; the load for this flight is 192 pounds. The two LEM RCS oxidizer tanks hold 552.4 pounds (see 33-1-11.14); the load for this flight is 383 pounds.

The 575 pounds of reactant is required for LEM attitude control (see 22-1-11.22).

4.2.2.4 Weight Profile

The vehicle weight changes for the portion of the mission beginning with S-IVB burnout through CM re-entry are shown in Figure 4-5. Data for this chart was taken from Reference 111, from the Weight Summary Chart (Table 4-1), and from the Expendables Weight Summary Charts (Figure 4-4).

Mission 4 vehicle weight at S-IVB burnout is 166,617 pounds. S-IVB staging and jettisoning of the LEM adapter reduces the weight to 110,000 pounds.

The payload weight required for this mission is 78,878 pounds. The differential between maximum vehicle weight and payload weight is shown as margin, which for this mission is 31,122 pounds.

During the 45-day earth orbit coast, the consumption of expendables reduces the spacecraft weight to 73,184 pounds.

LEM staging brings the CSM weight to 60,506 pounds at the beginning of retrofire. Fuel consumption for retro and SM staging then reduce the weight to 10,552 pounds, which is the CM re-entry weight.

4.2.3 CREW ANALYSIS

4.2.3.1 Time Utilization

Crew tasks performed during Mission 4 fall into two basic categories: flight operations and experiments. The crew flight operations required from liftoff to the 200-n. mi. orbit, and from initiation of de-orbit to touchdown, are detailed in Appendix D. Flight operations activities required to the crew during the 200-n. mi. orbit consist primarily of system status checks, IMU alignments, and communications with MSFN. Table 4-4 summarizes the system status checks adapted from the Apollo Design

Reference Mission (see 25). A detailed task analysis of the crew activities required for experiments is also presented in Appendix D.

Analysis of the flight crew operations, and the sleep, eating, hygiene and rest requirements for a typical orbital day, indicate that approximately 24.0 man-hours are available in a 24-hour day to perform experiments. This totals 1080 man-hours available for experiments over the 45 days. Utilizing the number of experiment sequences and the number of test subjects recommended for the assigned experiments (see 31, 32, and 33), with the experiment times as outlined in Appendix D, it was determined that approximately 1790 man-hours would be required to conduct the assigned experiments. Therefore, an experiment priority system was established.

Table 4-4

Breakdown of
Earth Orbital System Status Checks*

Task	Frequency of Performance	Duration (minutes)
CM		
Check ECS	Hourly	1
Check EPS	Hourly	3
Check SM RCS	Every 4 hours	4
Check CM RCS	Every 12 hours	4
Replace CO_2 and odor absorber	Every 12 hours	8
Charge entry and post landing batteries	Every 6 hours	3
Purge fuel cells	Every 8 hours	3
Check SPS	Every 4 hours	4
LEM		
Check ECS	Hourly	1
Check PGS	Hourly	1
Check PDS	Hourly	1
Replace CO_2 and odor absorber	Every 12 hours	8
Check LEM RCS	Every 12 hours	4
4.75 man-hours required for system status checks in a 24-hour day.		

*AMPTF adapted (24, 25, and 26).

CM tasks taken from translunar and transearth coast phase.

LEM tasks taken from Lunar Surface phase.

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WEIGHT PROFILE

MISSION 4 NEAR POLAR RETROGRADE SURVEY

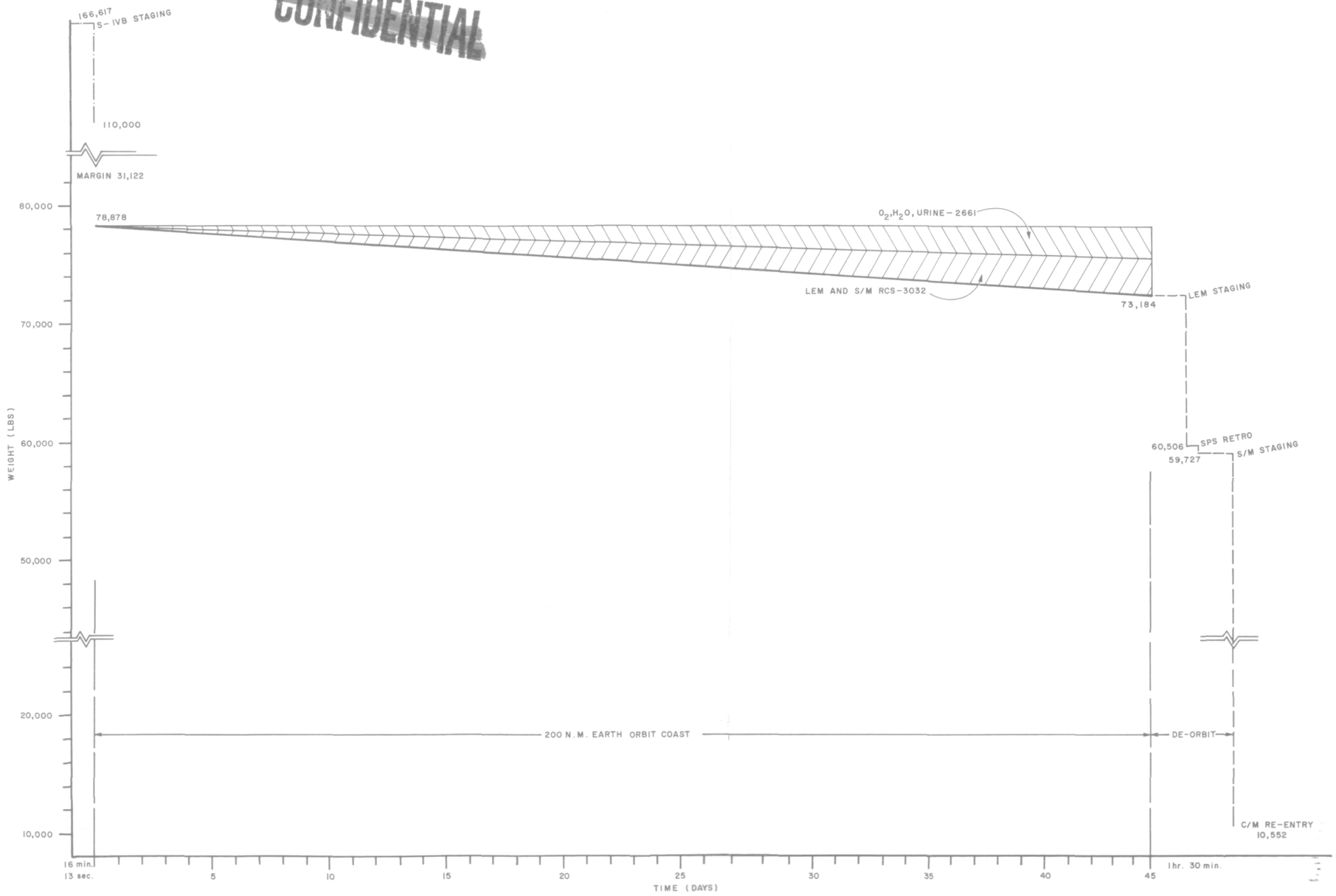

Figure 4-5. Mission 4 - Weights Profile

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4.2.3.2 Experiment Priorities

The priority classifications scheme and priority numbers are as follows:

a. First priority experiments are those that satisfy primary mission objectives.

b. Second priority experiments are those that contribute to the assessment of physical health or well being of the astronauts.

c. Third priority experiments are those that satisfy secondary mission objectives.

d. Fourth priority experiments are the biomedical and behavioral experiments, which contribute general scientific knowledge.

e. The lowest priority experiments are those that have been performed on previous flights, and are of such a nature that their omission is not detrimental to mission objectives.

It should be noted that the priority classification scheme and assignments were arbitrarily made because of the excessive experiment time load. They do not constitute a recommended or suggested solution to the experiment priority and integration problem.

Experiment priorities for the Mission 4 are as follows:

Priority	Experiment Number and Group	
-	1407	Onboard Guidance and Navigation (required for navigation calibration)
1	0900 Series	Survey - 0901, 0902, 0903
1	0802A, B, C, D	Star-Tracker and Infrared
2	0100 Series	Biomedical - 0109, 0110, 0111, 0112, 0113, 0114, 0118, 0115, 0116, 0119, 0120
2	0501	Radiation Monitoring
3	1405	Solar Cell
4	0100 Series	Biomedical - 0103, 0104, 0105, 0106, 0107, 0108, 0117
4	0200	Behavioral - 0201, 0202, 0203

4.2.3.3 Scheduling

The flight operations and experiment scheduling assumptions utilized in scheduling Mission 4 are as follows:

a. Experiment Scheduling - See Scheduling Consideration, Appendix D.

b. Flight Operation Scheduling

(1) Eight IMU alignments are required during a 24-hour period, five of which are required for experiments 0901/0902 (General Electric).

(2) Fourteen hours off duty per astronaut per day consisting of 6 hours sleep, 5 hours rest, and 3 hours eating and personal hygiene. Ten hours on duty (see 24, 25, and 26).

(3) System status checks for flight operations require 4.75 man-hours during a 24-hour period (AMPTF adapted, see 24, 25, and 26). Any system checks required for experiments are included in the experiment setup times.

Communication times for flight operations were not programmed on this flight because the contact points were not known at the time of scheduling.

The time utilization table, Table 4-5, shows the percentage of time actually scheduled for the four major crew activities and presents the total man-hours of experiments scheduled. As indicated by this chart, approximately 44 percent of the recommended experiment time was actually scheduled for the flight.

Table 4-5
Time Utilization

Mission	Time
Experiments	24.2 percent
Flight Operations	13.9 percent
Personal Time	57.4 percent
Unscheduled Time	4.5 percent
Experiment Time Recommended*	1790.0 man-hours
Experiment Time Scheduled	786.0 man-hours

*Grumman number for experiment sequences and subjects tested (see 31, 32, and 33), experiment times from Appendix D.

The actual schedules for the first 14 days of the mission are presented in Figure 4-6. This 14-day sample presents the pattern of experiment sequences that can be expected for a complete 45 days. Table 4-6 provides the total amount of time scheduled for each experiment, the number of experiment sequences actually scheduled, and the number recommended (31, 32, and 33) for a 45 day flight.

Table 4-6

Flight 518 - Summary of Scheduled Experiment Times and Sequences
(14 Day Mission Sample)

Experiment Number	Man-Minutes Expended (14 days)	Frequency of Performance	
		Actual (14 days)	Recommended (45 days)
0103	292	2	12
0104	400	2	12
0105	214	1	6
0106	312	4	17
0107	368	4	17
0108	192	2	9
0109	404	2	15
0110			
0111	164	2	9
0112	46	1	9
0113	428	3	5
0114	54	2	36
0115	240	1	18
0116			
0117A	325	13	45
0117B	40	2	6
0118	96	2	5
0119	180	1	5
0120	68	1	9
0121	-	0	2
0201A	86	1	15
0201B			7
0201C	-	0	15
0201D	-	0	7

Table 4-6

Flight 518 - Summary of Scheduled Experiment Times and Sequences
(14 Day Mission Sample) (Cont.)

Experiment Number	Man-Minutes Expended (14 days)	Frequency of Performance	
		Actual (14 days)	Recommended* (45 days)
0201E	76	1	7
0202A	110	1	15
0202B			
0202C	-	0	15
0202D	-	0	15
0202E	-	0	15
0202F	-	0	15
0202G	-	0	15
0202H	-	0	15
0202I	-	0	15
0203A	41	1	15
0203B	48	1	15
0203C	38	2	15
0203D	87	2	7
0203E	105	1	7
0203F			15
0203G			15
0203H			15
0501	520	40	90
1405	1381	12	43
0802A	3111	51	180
0802B			180
0802C			90
0802D	18	1	6
0901	2676	61	Continuous **
0902			Continuous **
0903	774	10	Continuous **
1407A	530	10	25
1407B	550	10	25
1407C	550	10	25

* See (33-1.11).
**See Experiment Assumption No. 1.

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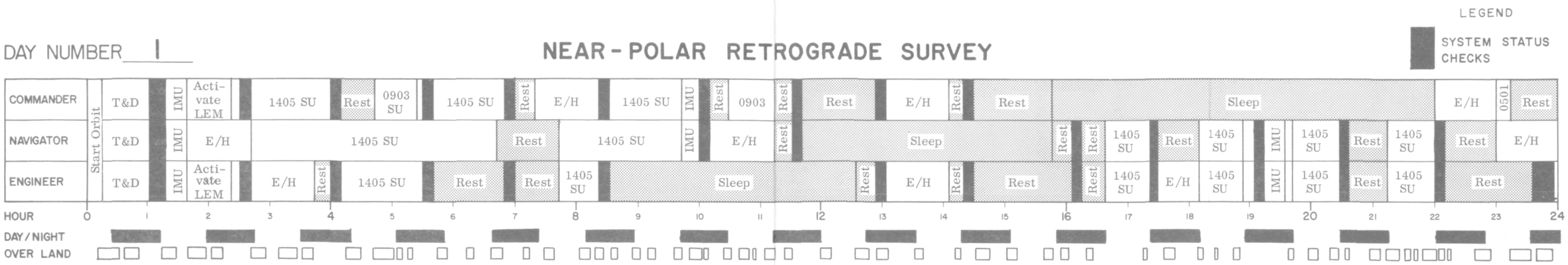
MISSION NUMBER 4

DAY NUMBER 1

NEAR-POLAR RETROGRADE SURVEY



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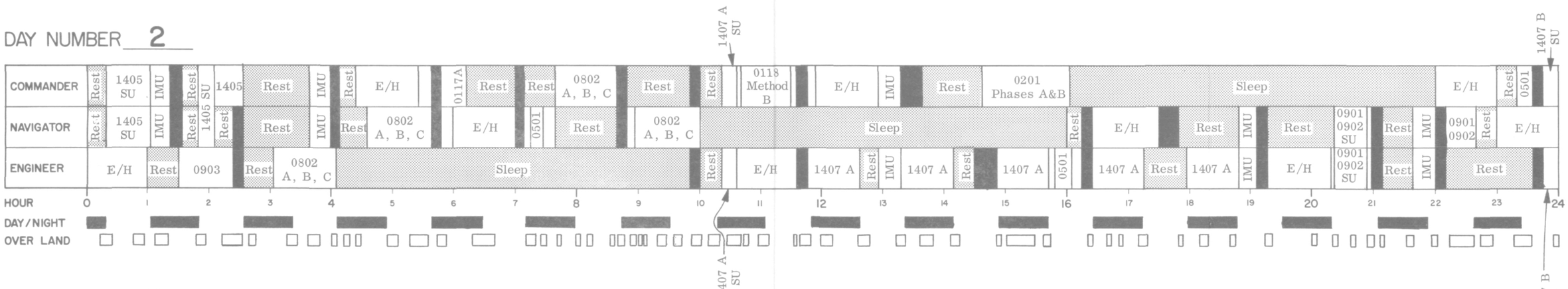

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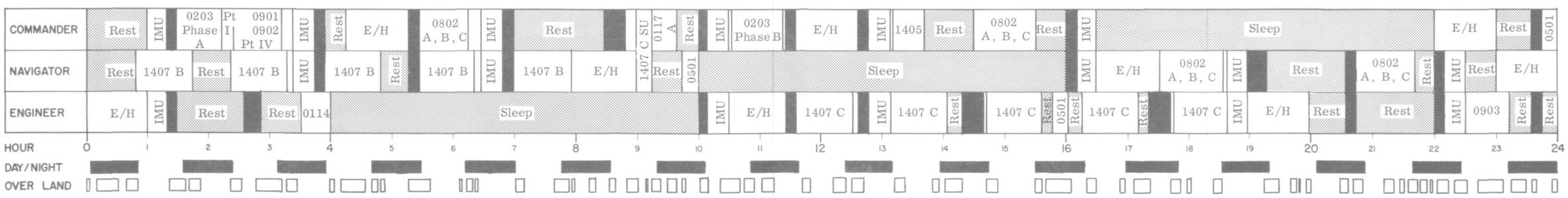

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Figure 4-6. Integrated Mission Schedule (Sheet 1 of 5)

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MISSION NUMBER 4

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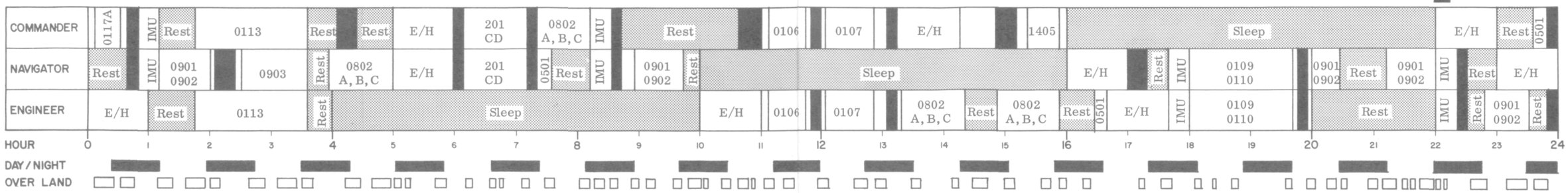
NEAR–POLAR RETROGRADE SURVEY

LEGEND
SYSTEM STATUS CHECKS

DAY NUMBER 4



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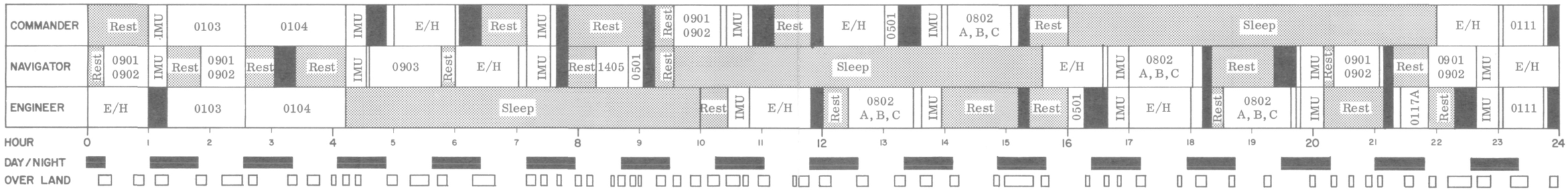

DAY NUMBER 6

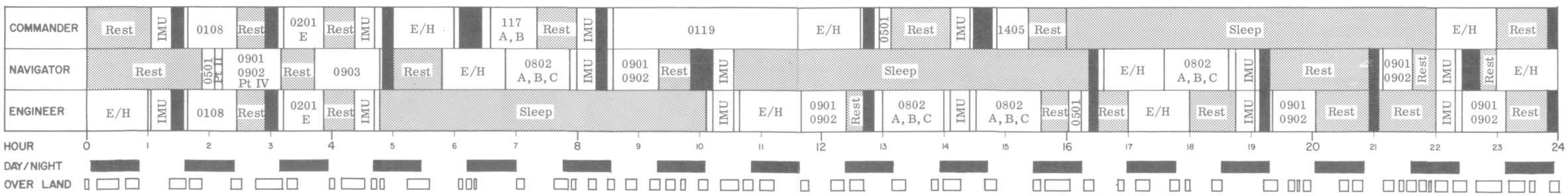

Figure 4-6. Integrated Mission Schedule (Sheet 2 of 5)

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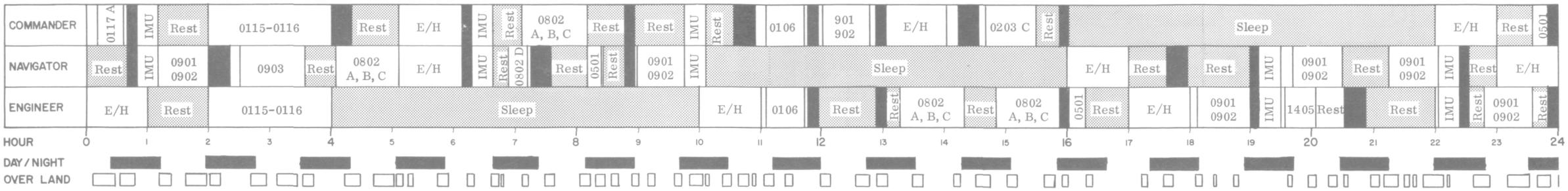
MISSION NUMBER 4

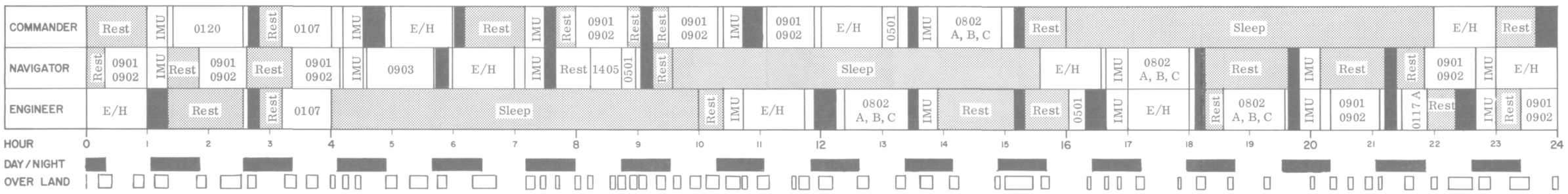
NEAR–POLAR RETROGRADE SURVEY

LEGEND
SYSTEM STATUS CHECKS

DAY NUMBER 7



DAY NUMBER 8



DAY NUMBER 9

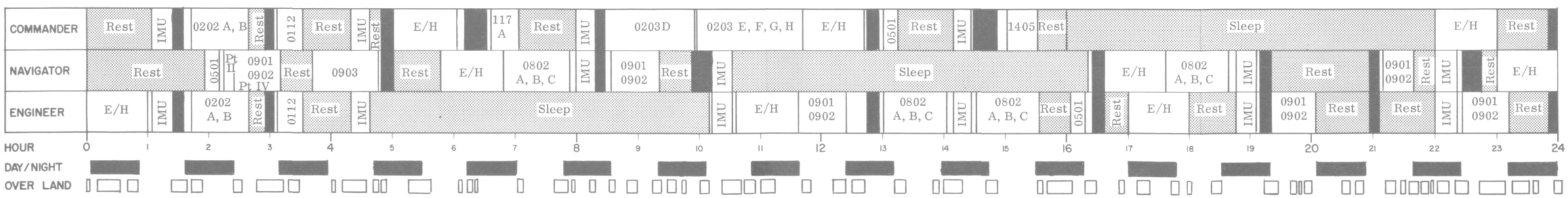


Figure 4-6. Integrated Mission Schedule (Sheet 3 of 5)

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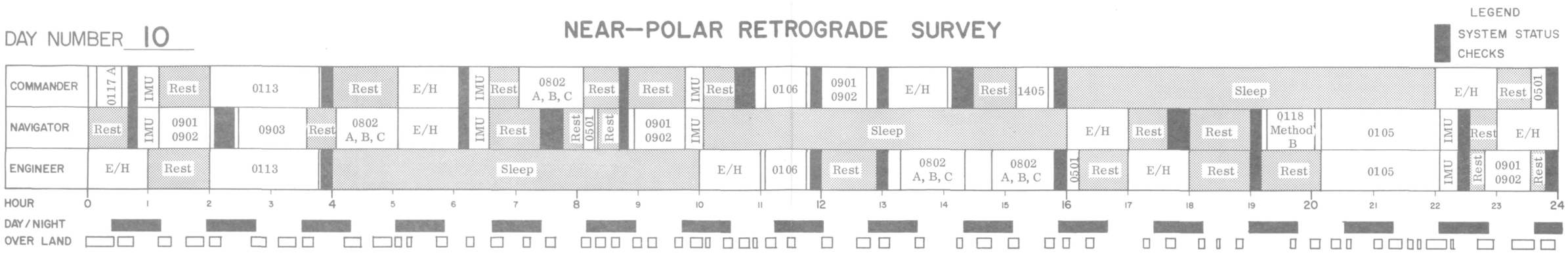
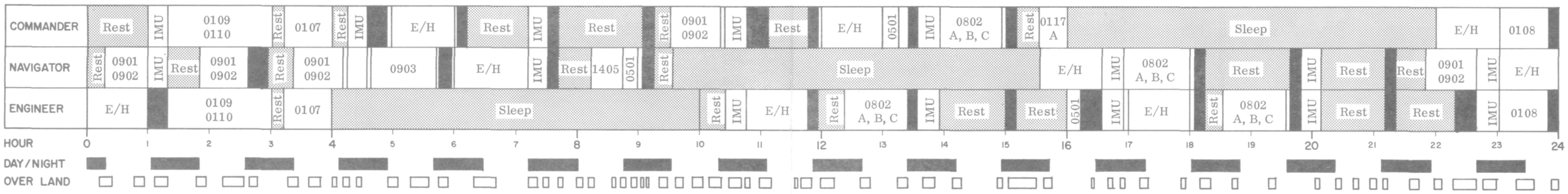
MISSION NUMBER 4

DAY NUMBER 10

NEAR–POLAR RETROGRADE SURVEY



DAY NUMBER 11



DAY NUMBER 12

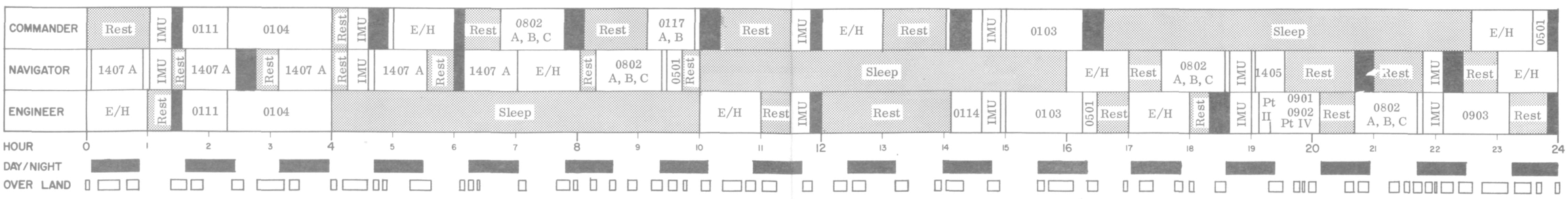


Figure 4-6. Integrated Mission Schedule (Sheet 4 of 5)

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MISSION NUMBER 4

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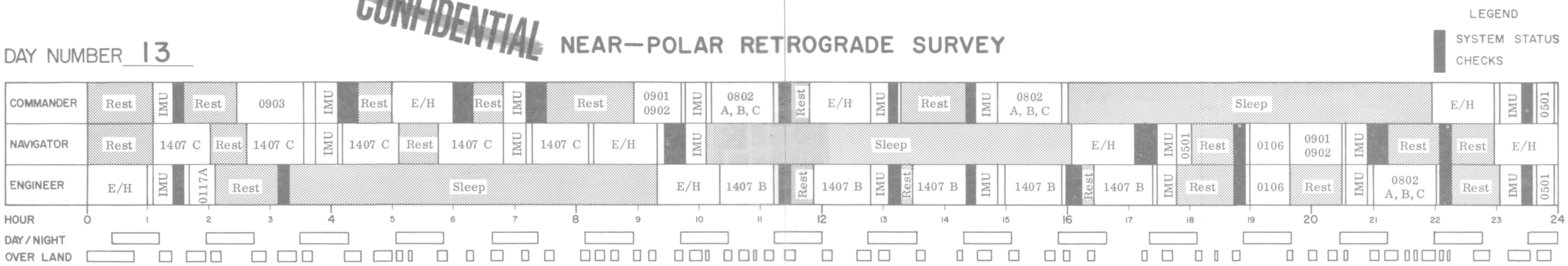
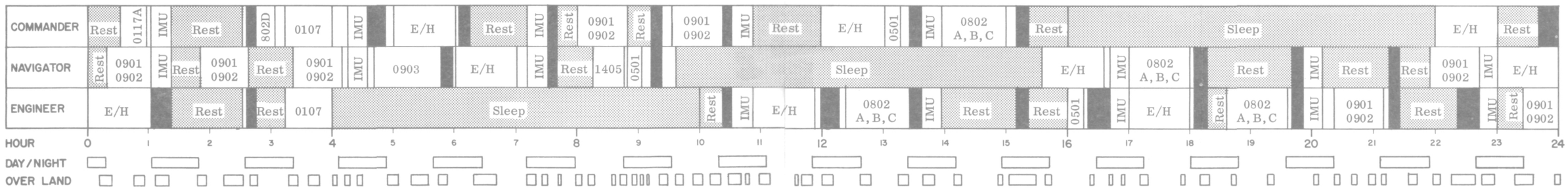
NEAR—POLAR RETROGRADE SURVEY

DAY NUMBER 13



DAY NUMBER 14



DAY NUMBER _____

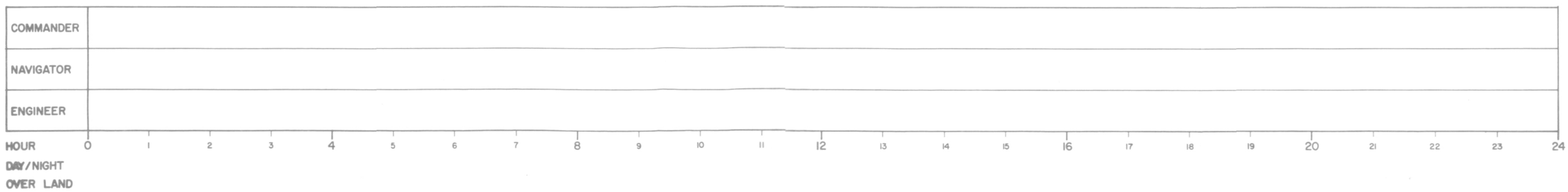


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Figure 4-6. Integrated Mission Schedule (Sheet 5 of 5)

4-41-1

4.2.4 DATA FLOW

4.2.4.1 Introduction

Data flow characteristics - data recording and transmission, magnetic tape storage, and data collection - are detailed in the following paragraphs. The data flow was profiled for Mission 4 using the analysis presented in Part II, Section 3, of this report. Instructions for using the profiles are included in the section under the applicable heading.

4.2.4.2 Data Recording and Transmission

A data recording and transmission profile for the first three days of Mission 4 are presented in Figure 4-7. A sample profile was developed for each mission to illustrate graphically the distribution of recording and transmission times.

The top line shows the distribution of experiment data collection sequences and their duration in time. The second line shows the distribution of MSFN S-band telemetry ground station coverages and their durations. The third line shows the coverage distribution of those telemetry stations assumed to have television receiving capability (Goldstone, Madrid, and Canberra). The fourth line shows the distribution of station contacts tentatively assigned for the reception of dumped operational data (and low-sampling-rate experimental data) from the CSM recorder.

NOTE

These contacts were assigned on a one-dump-per-orbit basis with no interval greater than 120 minutes whenever possible. When intervals greater than 120 minutes occurred, an extra operational contact was tentatively assigned to allow the dumping of the extra reel of stored operational data tape. Whenever a choice was available, the operational contact was made over the U.S.

The fifth line is a profile of the rate of accumulation of the magnetic tape being used to record high rate PCM data in the LEM, compared to the rate at which reels of this tape may be dumped to the ground telemetry stations. Each upward-pointing arrow indicates the accumulation of one full reel of tape (30 minutes recording time) and each downward-pointing arrow indicates the completed dump of one full reel of tape (30 minutes of dumping time).

The recording and transmission profile for the first three days of Mission 4 is not typical of any three-day period later in the mission. The later recording profile shown in the mission reveals a very high rate of recording as evidenced by the excess magnetic tape recovery requirements given in the Data Collection Summary Chart, Table 4-7.

It is clearly evident from the recording and transmission profile that a great amount of available transmission time is wasted, particularly on the first day. This is because no recording tape backlog is presently scheduled to be accumulated before starting the lengthy solar cell deployment procedure in experiment 1405. This experiment should be rescheduled for later in the mission.

During the generation of the telemetry coverage profile, it was observed that the addition of the Fairbanks, Alaska, station raised the coverage from a level of 11 percent to a level of 14 percent of the total orbital time. It also reduced the occurrence of excessive noncoverage intervals (intervals greater than 120 minutes) from an average of two such intervals per day to an average of less than one every three days.

4.2.4.3 Magnetic Tape Storage Profile

For each applicable mission a sample magnetic tape storage profile was developed for the first three days of that mission to show the nature of the net accumulation of reels of recorded magnetic tape. This profile is derived directly from the fifth line on the Data Recording and Transmission Profile. Figure 4-8 is the tape storage profile for Mission 4.

This profile is also not typical of that obtained later in the mission, when reels of excess tape are accumulated at the rate of approximately 8 reels per day. This rate of accumulation is a serious problem, particularly in view of the fact that mapping film is also being rapidly accumulated during the course of the mission.

4.2.4.4 Data Collection Summary

The data collection requirements for Mission 4 are summarized in Table 4-7. The first column designates each experiment by experiment number (see 38, 39, 40, and 77). The exception to this is series 0100, the biomedical experiments, and series 0200, behavioral experiments. The next column is the maximum number of special analog channels that must be recorded at any one time due to that experiment. The next indicates whether or not the digital output of a PCM multiplexer must be recorded. The

Table 4-7

Data Collection Summary

Experiment	Data Recording							Recovery Requirements				Remarks
	Max. No. of Channels in Use at One Time		Minutes per Sequence	Total No. of Sequences	Total Minutes Recorded	Transmission Time Avail. (Minutes)	Recorded Back Log (Minutes)	Magnetic Tape		Pounds of Film	Miscellaneous	
	Anal.	Dig.						Reels	Lbs.			
	*	*				**	***	**	**			* Assumed time sharing of channels as required. ** Summed only for total mission. ***Difference - Recorded minus Transmitted - only for total mission.
0100	6	1	*	111**	3904					X	X	* Varies for each experiment in series; see Data Collection Charts, Part II, Section 3. ** Varies for each experiment in series; number (111) is total for series for which recorder is used.
0200	1	1	*	18**	1410					145	X	* Varies for each experiment in series; see Data Collection Charts, Part II, Section 3. ** Varies for each experiment in series; number (18) is total for series for which recorder is used.
												X Includes a requirement - no data or rate.

Table 4-7

Data Collection Summary (Cont.)

Experiment	Data Recording							Recovery Requirements				Remarks
	Max. No. of Channels in Use at One Time		Minutes per Sequence	Total No. of Sequences	Total Minutes Recorded	Transmission Time Avail. (Minutes)	Recorded Back Log (Minutes)	Magnetic Tape		Pounds of Film	Miscellaneous	
	Anal.	Dig.						Reels	Lbs.			
0501		1	2	40	80					X		
0802A, D		2	25	168	4200					30		
0901	6	2	40*	210	8400					905		* Average minutes per sequence, estimated.
0902	6	2	40*	210	8400					905		* Average minutes per sequence, estimated.
0903		1	40	45	1800					370		
1405		1	30	45	1500					X*	X**	* 1000 feet, 16 mm. **100 feet, 35 mm still shots.
1407		1	20		1500					X*		* 1000 feet of film
Total	6*	3*			19794**	9404	10390	347	694	1450	X	* Total number of channels in use at one time during mission, not a column summation. Assumes time sharing as required. **1405 and 1407 not included, low rate data only. X Includes a requirement - no data or rate.

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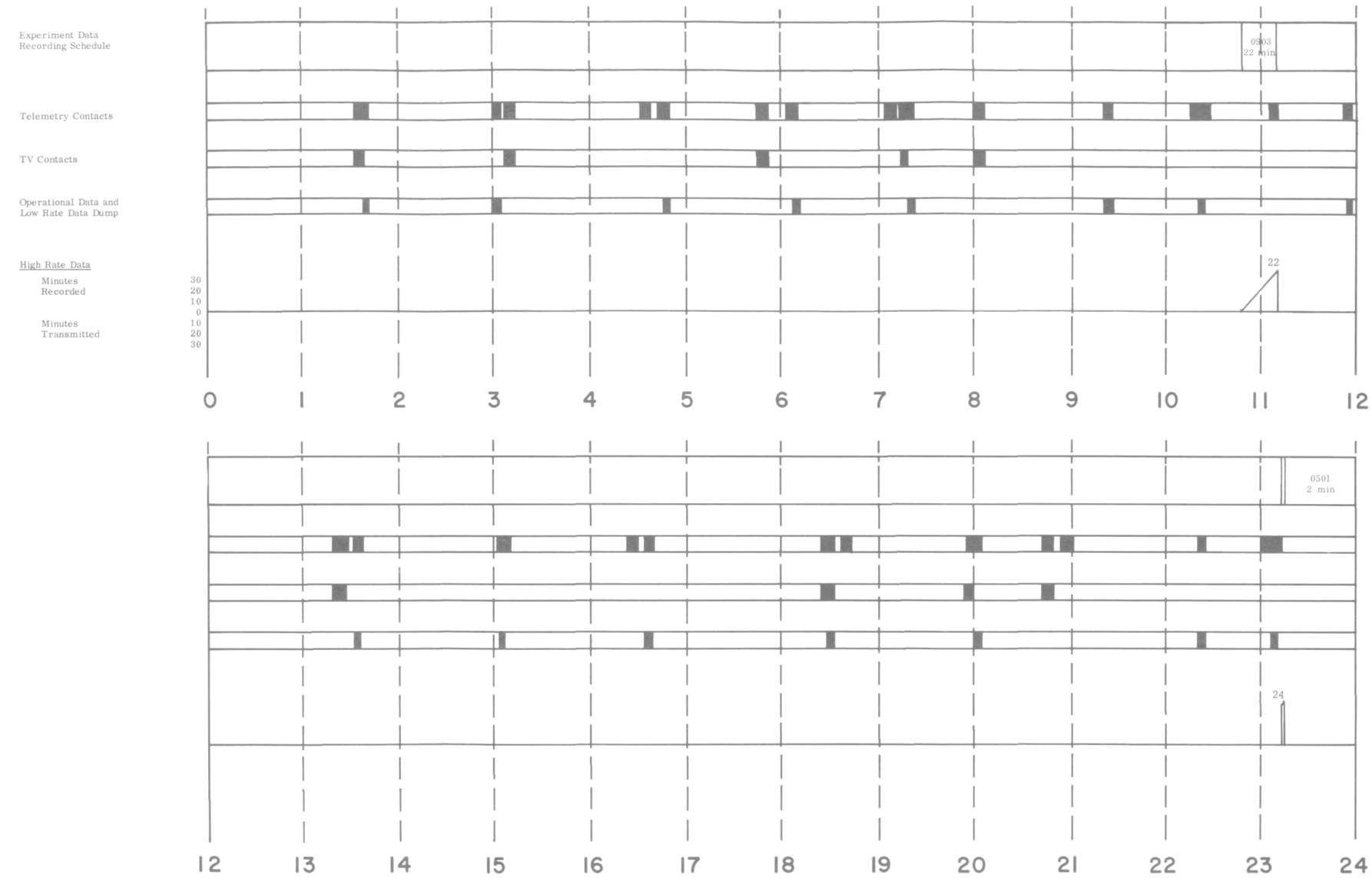
MISSION 4 DAY 1



DATA RECORDING AND TRANSMISSION PROFILE

Figure 4-7. Data Recording and Transmission Profile (Sheet 1 of 3)

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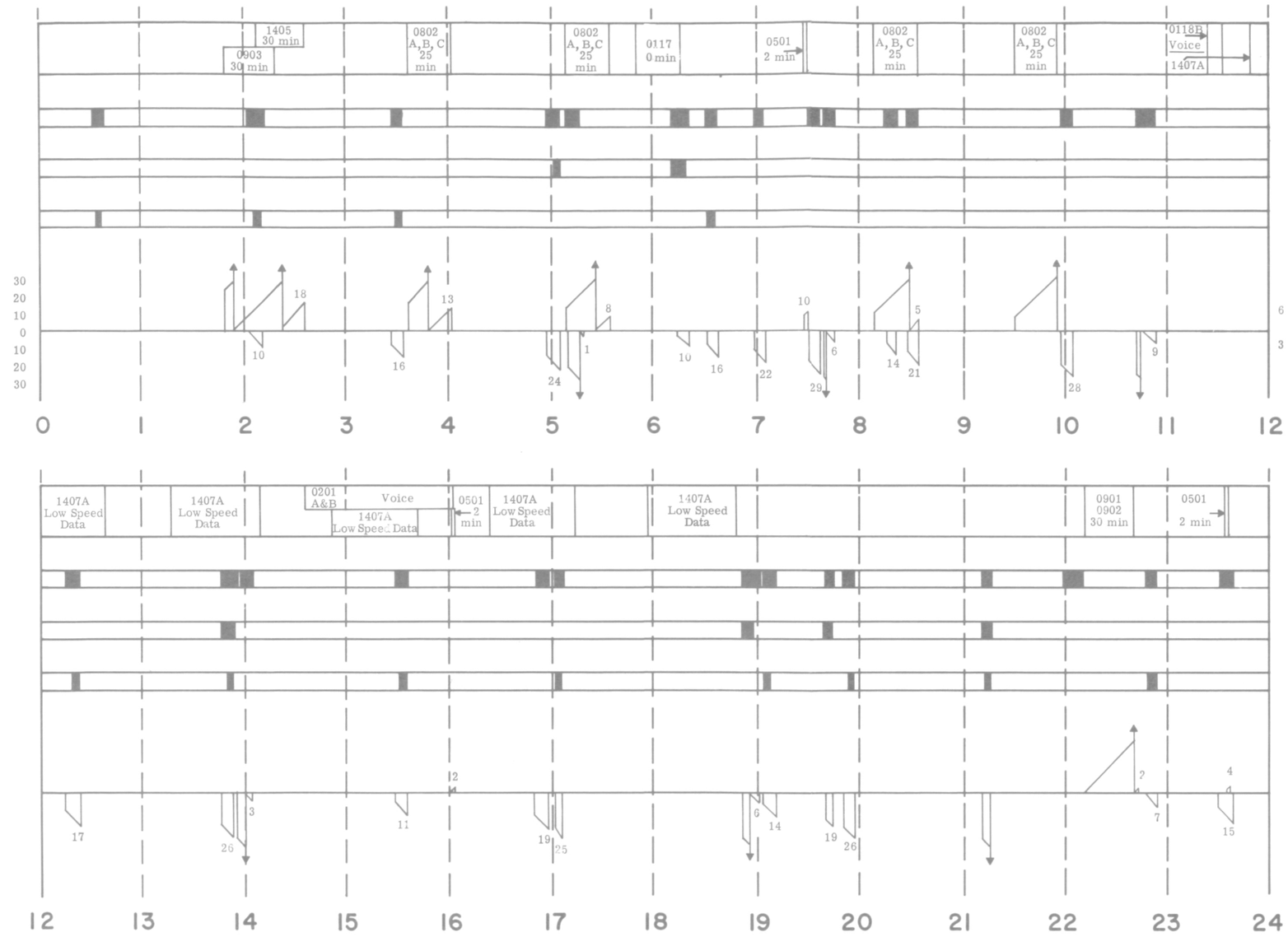
MISSION 4 DAY 2



DATA RECORDING AND TRANSMISSION PROFILE

Figure 4-7. Data Recording and Transmission Profile (Sheet 2 of 3)

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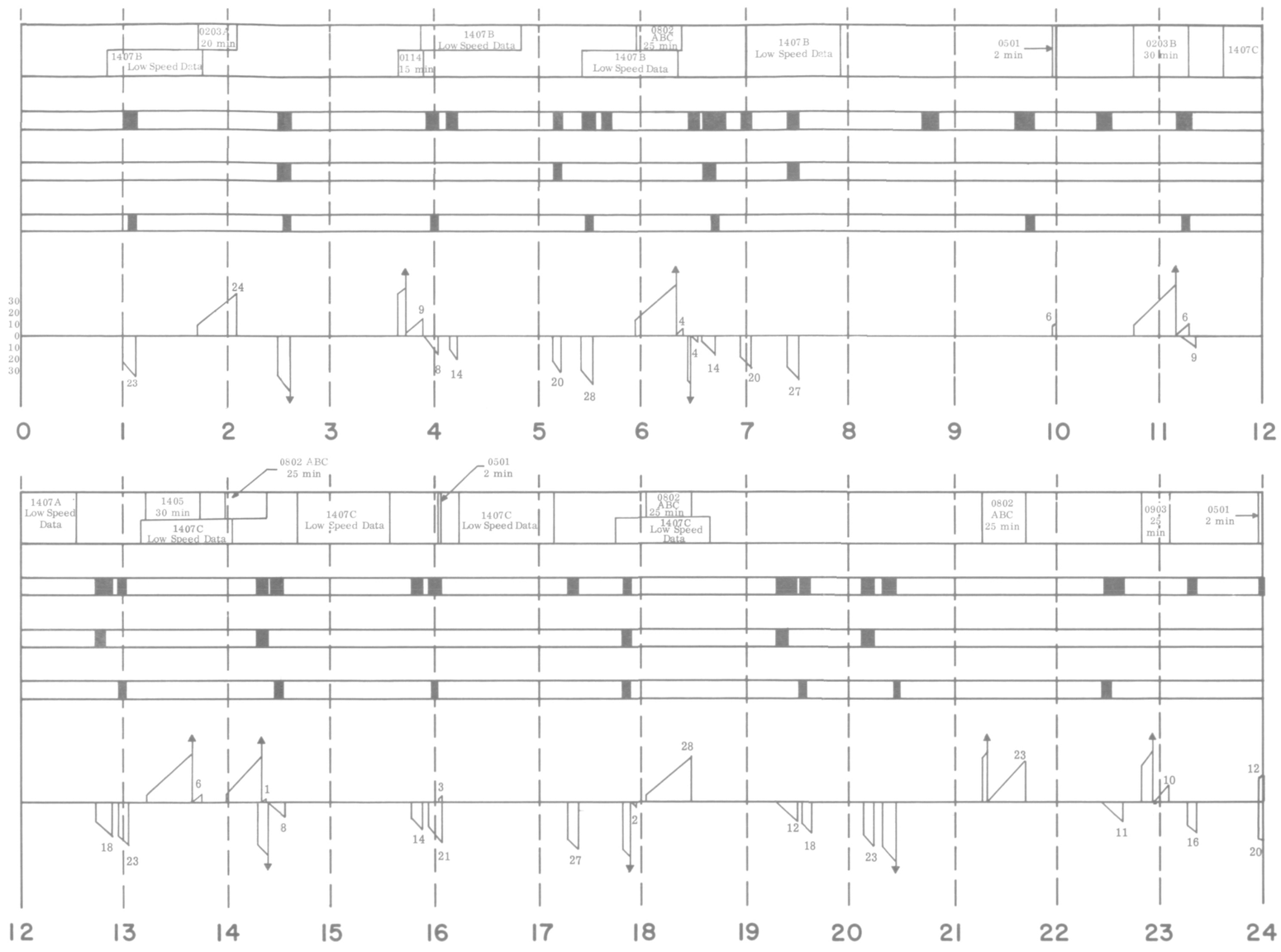
MISSION 4 DAY 3



DATA RECORDING AND TRANSMISSION PROFILE

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Figure 4-7. Data Recording and Transmission Profile (Sheet 3 of 3)

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MAGNETIC TAPE STORAGE PROFILE
MISSION 4

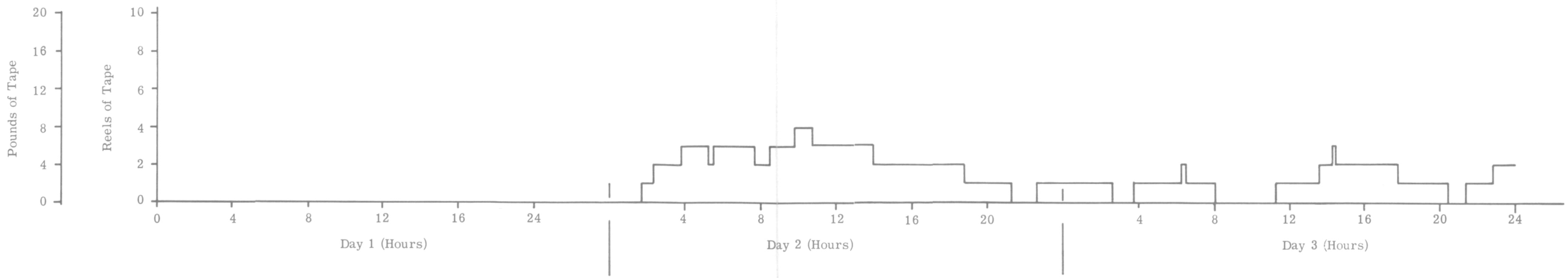

Figure 4-8. Magnetic Tape Storage Profile

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fourth column indicates the minutes of recording time required by each experiment sequence. The fifth and sixth columns indicate the total number of sequences and total minutes of recording time, respectively, required by that experiment during the course of the mission.

The next column contains a graphically derived value of the total transmission time available during the mission and is entered only in the over-all totals. The figure given for "Recorded Backlog" is the difference between the over-all total "high rate" PCM and analog data recording time and the over-all total available transmission time. Further experiment data collection details are given in Part II, Section 3.

The chart for Mission 4 indicates a requirement for a recoverable payload of over 2144 pounds, which is eight times greater than the assumed recoverable payload capability. Several other contributions to the required payload are required for which no reasonable estimates are available. The data collection requirements for this mission require further study. This problem is discussed in Part II, Section 3.

4.2.5 TRAJECTORY ANALYSIS

4.2.5.1 General

This mission utilizes a Saturn V launch vehicle to place an experiment package into a 200-n.mi., 96.5 degree inclination orbit. The launch phase crosses the southern coast of Florida with a launch azimuth of 146 degrees from Cape Kennedy. The yaw program of 1-degree per second is used between 120 to 150 and 159 to 183.8 seconds to achieve a direct ascent into mission orbit. The S-IC, S-II, and all but 16,078 of 230,972 pounds of S-IVB propellants are utilized (see 49, Table 13). The selected re-entry trajectory utilizes the south-moving orbit to obtain tracking by the station to be located in Fairbanks, Alaska. A chart showing early mission orbit ascending nodes is included in Volume IV. The ground track for the corresponding mission orbits may be found by using the appropriate transparent overlays.

4.2.5.2 Assumptions

The following assumptions were used in this study:

a. The launch time is established at 14:40 GMT, 1 October 1970.

b. A 45-degree upper limit was assumed in choosing an acceptable sun elevation angle at launch for photographic purposes. This requirement established the launch time.

c. A 0-n.mi. vacuum perigee re-entry trajectory was assumed.

4.2.5.3 Analysis

One of the constraints on launch time is the desired 45 degree angle to the sun (see 35), for operation of the survey camera. The lighting conditions existing throughout the mission may be determined by using the transparent illumination overlays provided in Volume IV. Instructions for using these charts are included in that volume. The 96.5-degree inclination orbit is a retrograde orbit (velocity component opposite the earth rotation direction - angular momentum vector in the southern hemisphere), that has a 1-degree per day easterly precession. This precession of the orbit in inertial space compensates for the movement in longitude of the subsolar point caused by the yearly movement of the earth around the sun. Since desired or acceptable ranges of sun lighting were not specified, the lower angles (from sun to local horizontal) are assumed more desirable, in a manner analogous to the lunar polar survey mission. Thus, the orbit is located tangent to the inner lighting ring "limit" of 45 degrees. Using a -3-degree sun declination, a 15-degree sun angle will occur at a latitude of 65.5 degrees north, increase to a sun angle of 45 degrees near the equator,

and decrease to 15 degrees at a latitude of 71.5 degrees south. As the sun declination advances during the mission toward its maximum value of 23.5 degrees, the 45-degree elevation contours will lie at more southerly latitudes.

Since the mission orbit insertion longitude is 89 degrees west and the desired solar subpoint is 44 degrees west, this longitude corresponds to 14:56 GMT orbit insertion time using 4 minutes per degree. With a 16-minute launch-to-orbit insertion time, the launch time of 14:40 GMT or 9:40 EST is established. No acceptable camera lighting angle tolerance was specified in Reference 35, and a $\pm$ 18.8-degree range about 45 degrees corresponding to a two-and-half-hour launch window may be unacceptable.

The mission altitude of 200 n.mi. was based on a pessimistic atmosphere model for a lifetime of 45 days and a 1-week safety factor (see 2). The launch to orbit insertion phase of the trajectory is shown in Figure 4-9. Since the S-IC powered flight of the trajectory presented in Reference 49 passes over southern Florida, it is not possible at this time to give the location of the impact points of the jettisoned S-IC and LES equipment.

The experiment flight mechanics study yielded the following additional results:

a. Seasonal requirements - midsummer/ midwinter (see 1).
b. Vehicle orientation - earth pointing (see 35).
c. Stabilization requirements - $\pm$ 0.2 degree (540 hours) and $\pm$ 5 degrees (540 hours) (see 35).
d. Ephemeris accuracy requirements - 10 feet (see 35).

A detailed listing of flight dynamics parameters for all powered phases of the mission is given in Table 4-8.

Data for the launch from liftoff through injection into the 200-n. mi. circular orbit is taken from Reference 49. Computation supplied the remainder of the data. The launch and re-entry traces for vehicle B are shown on a uniform-grid projection in Figure 4-10. Ground tracking coverage and illumination for the ascent phases are given in Figure 4-11.

The ground traces for the later orbits can be obtained using the uniform-grid projection with ascending nodes plotted on it in conjunction with transparent overlays. These materials are included in Volume IV of this report.

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Table 4-8

Saturn V Reference Trajectories, Low-Earth Orbits
(200 N. Mi. Circular, i = 96.5 Degrees, S-IVB Stack Limit)

Trajectory Event	Time (Sec)	Geodetic Latitude (Degree)	Longitude (Degree)	Velocity(1) (Ft/Sec)	Altitude (Ft)	Flight(1) Path Angle (Degree)	Azimuth(2) (Degree)	Vehicle Weight (Lb)
1. Liftoff	0.00	28.65	-80.64	1,340	0	0.00	-----	6,367,048
2. End Vertical Rise, Begin Zero-Lift Flight	12.00	28.65	-80.64	1,343	505	3.79	146.0	6,020,588
3. Shutdown of S-IC Inboard Engine	146.57	28.42	-80.50	5,948	227,908	52.88	161.8	2,135,403
4. Shutdown of S-IC Outboard Engines (3), Begin Coast	150.57	28.39	-80.48	6,215	247,244	52.65	164.6	2,043,013
5. Jettison of S-IC, SII Ignition, Begin Pitch-up Maneuver (4), Mixture Ratio = 5.0	154.37	28.36	-80.47	6,121	265,764	52.01	164.6	1,430,394(5)
6. End Pitch-up Maneuver, Change Mixture Ratio to 5.4	164.37	28.27	-80.45	6,069	313,260	50.67	166.4	1,406,098
7. Jettison S-IC/S-II Interstage Adapter Section	180.57	28.13	-80.42	5,951	387,588	48.64	170.5	1,352,888(6)
8. Jettison Launch Escape System	185.57	28.08	-80.41	5,945	409,794	47.90	172.0	1,331,281(7)
9. Change Mixture Ratio to 4.7	378.28	25.22	-81.06	8,633	1,008,434	14.12	199.2	814,520
10. Shutdown of S-II, Begin Coast	526.19	20.99	-82.74	14,862	1,249,620	6.05	199.9	482,432
11. Jettison S-II, S-IVB Ignition	530.99	20.81	-82.81	14,848	1,256,776	5.70	199.9	381,511(8)
12. Burnout S-IVB, End Integrated Trajectory	973.23	-3.19	-89.02	25,228	1,215,772	0.00	186.5(1)	166,617(9)
13. SM Ignition for Deboost Maneuver	0.00	0.00	35.00	25,228	1,215,772	0.00	6.5(1)	22,432
14. Service Module Burnout	12.50	0.00	35.00	24,869	1,215,383	0.00	6.5	21,571
15. Command Module Re-entry	1,704.10	+66.29	-137.1	25,831	400,000	-1.52	196.3	10,303

(1) Inertial quantities.
(2) Relative quantities.
(3) S-IC was shutdown 8 seconds early to place spent booster and LES in a safe impact area; however, the trajectory reported here apparently does not accomplish this.
(4) Pitch-up of one-deg/sec for 10 seconds approximates optimum trajectory after which a pitch-down rate of 0.0871 degree/second is maintained to S-IVB burnout.
A yaw program of one-degree/second between 120 and 150 and between 159 and 183.8 seconds provides an orbit inclination of 96.5 degrees.
(5) After jettison event of 612,619 pounds.
(6) After jettison event of 9,770 pounds.
(7) After jettison event of 8,200 pounds.
(8) After jettison event of 100,921 pounds.
(9) Represents unconsumed S-IVB propellant (16,078 pounds), S-IVB jettison weight (40,539 pounds), and 110,000 pounds.

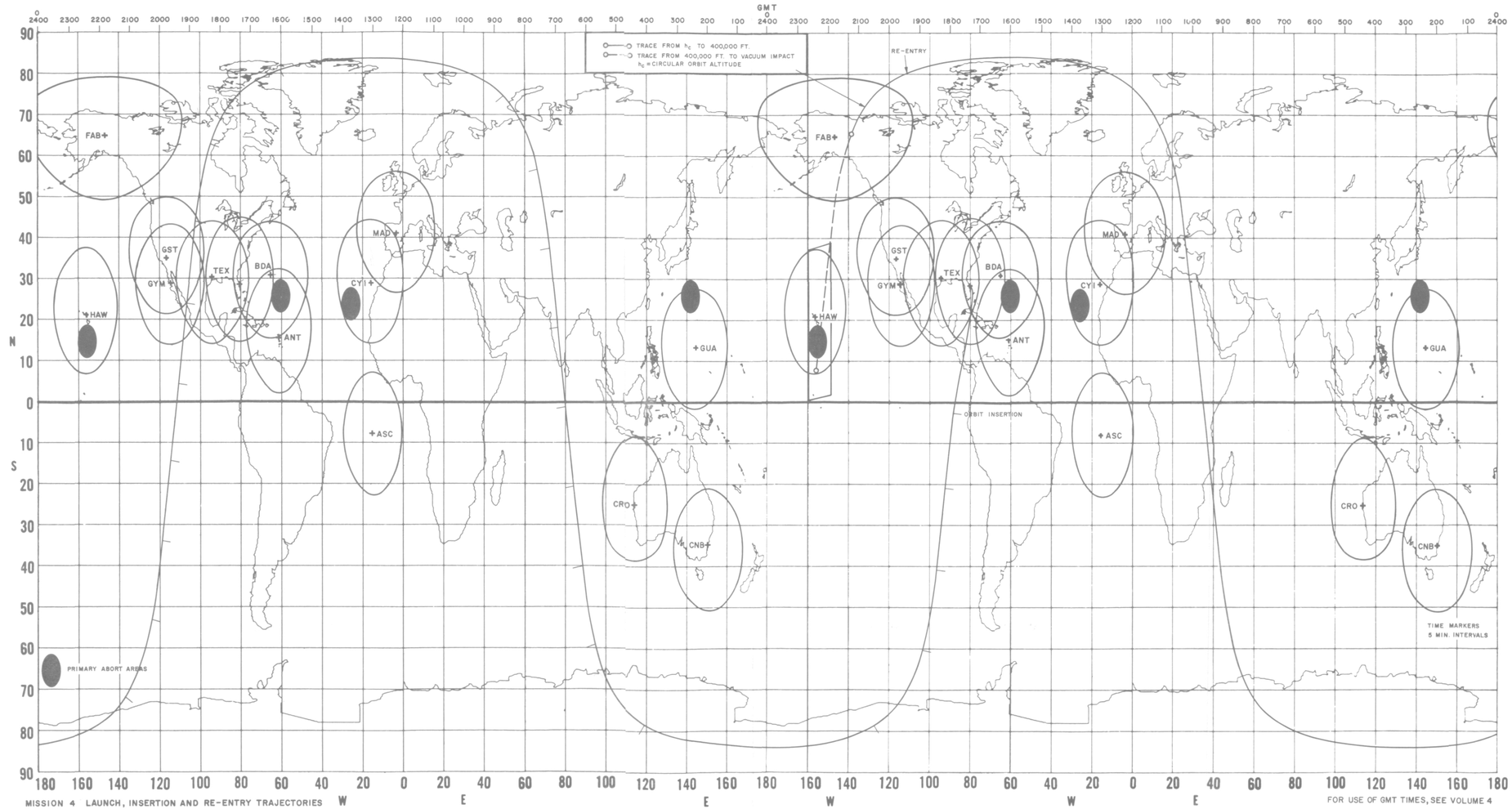

Figure 4-9. Launch and Insertion into Mission Orbit Ground Trace

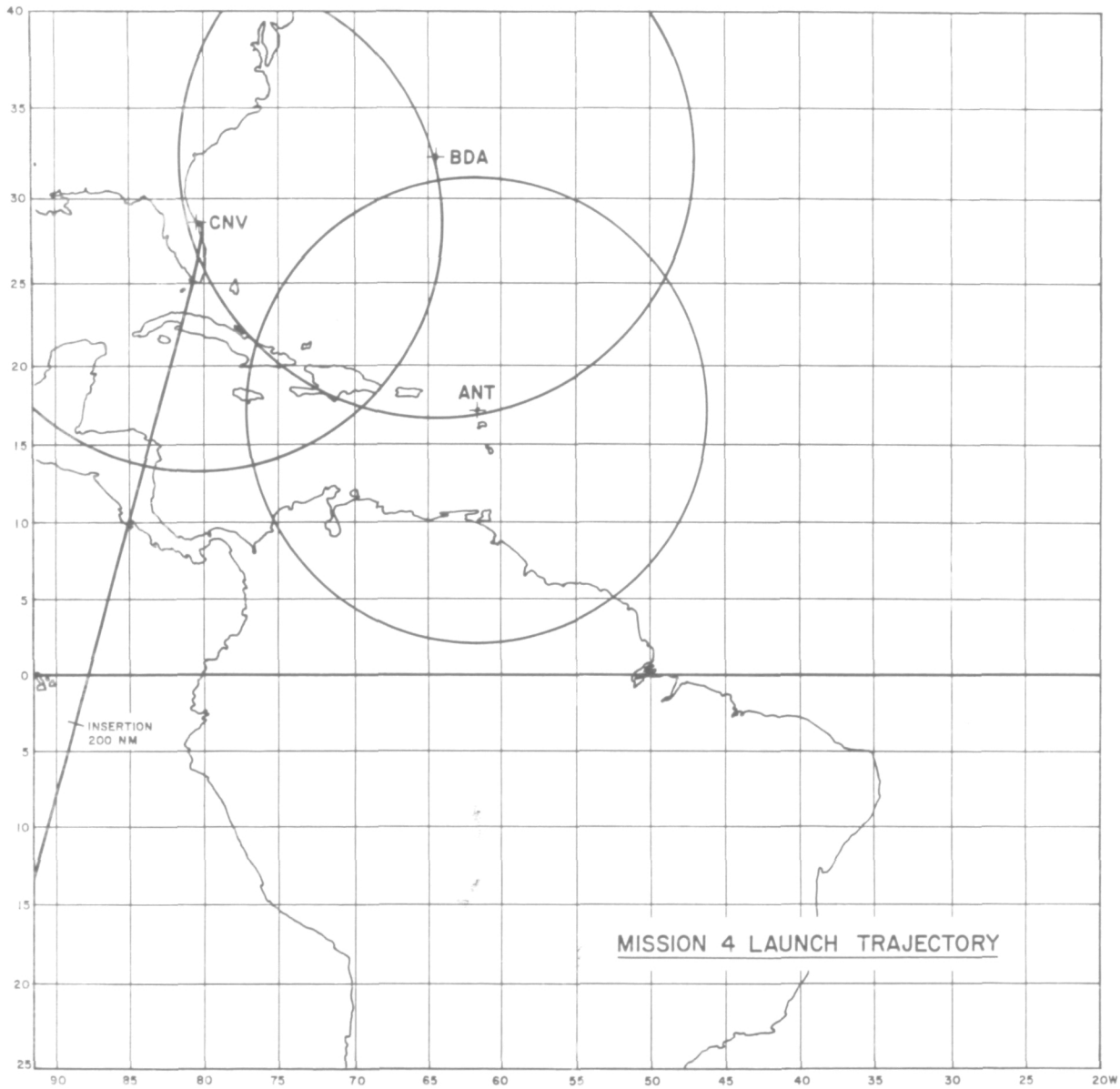

Figure 4-10. Uniform Grid Chart Ground Trace Launch Phase and Re-entry (See Reference 111)

4.2.6 ABORT ANALYSIS

4.2.6.1 General

Four major abort modes are applicable to Mission 4 (see 24, 25, 26, 105, 106 and 107):

a. LES Abort.
b. Suborbital abort.
c. Abort to orbit.
d. De-orbit abort.

The relationship between the mission phases and the abort mode available at any particular point within the phase is shown in Figure 4-12.

Ground rules for mode descriptions and abort logic diagrams for each abort mode are presented in the following paragraphs.

4.2.6.2 Launch Escape System Aborts

4.2.6.2.1 Ground Rules

A LES abort can be initiated anytime between LES enablement during prelaunch and LES jettison during the second stage thrusting period. LES is the primary abort mode from the time of LES enablement to the time LES is jettisoned. Launch vehicle engine shutdown must precede LES ignition for all aborts except those initiated during the first 40 seconds of flight. During the first 40 seconds, engine shutdown is inhibited by an AMR range safety constraint.

An automatic sequencing of spacecraft events is provided for LES aborts initiated below an altitude of 120,000 feet. In aborts initiated above 120,000 feet, the astronauts must jettison the LES manually, after escape motor thrust tail-off, and must orient the spacecraft to a heat-shield-forward attitude.

4.2.6.2.2 Mode Description

The operational events that follow the initiation of LES abort are shown in Figure 4-13. The sequence is automatically controlled by sequential timers and barometric switches.

MISSION 4 - NEAR POLAR RETROGRADE SURVEY

ILLUMINATION AND RADAR COVERAGE CHART

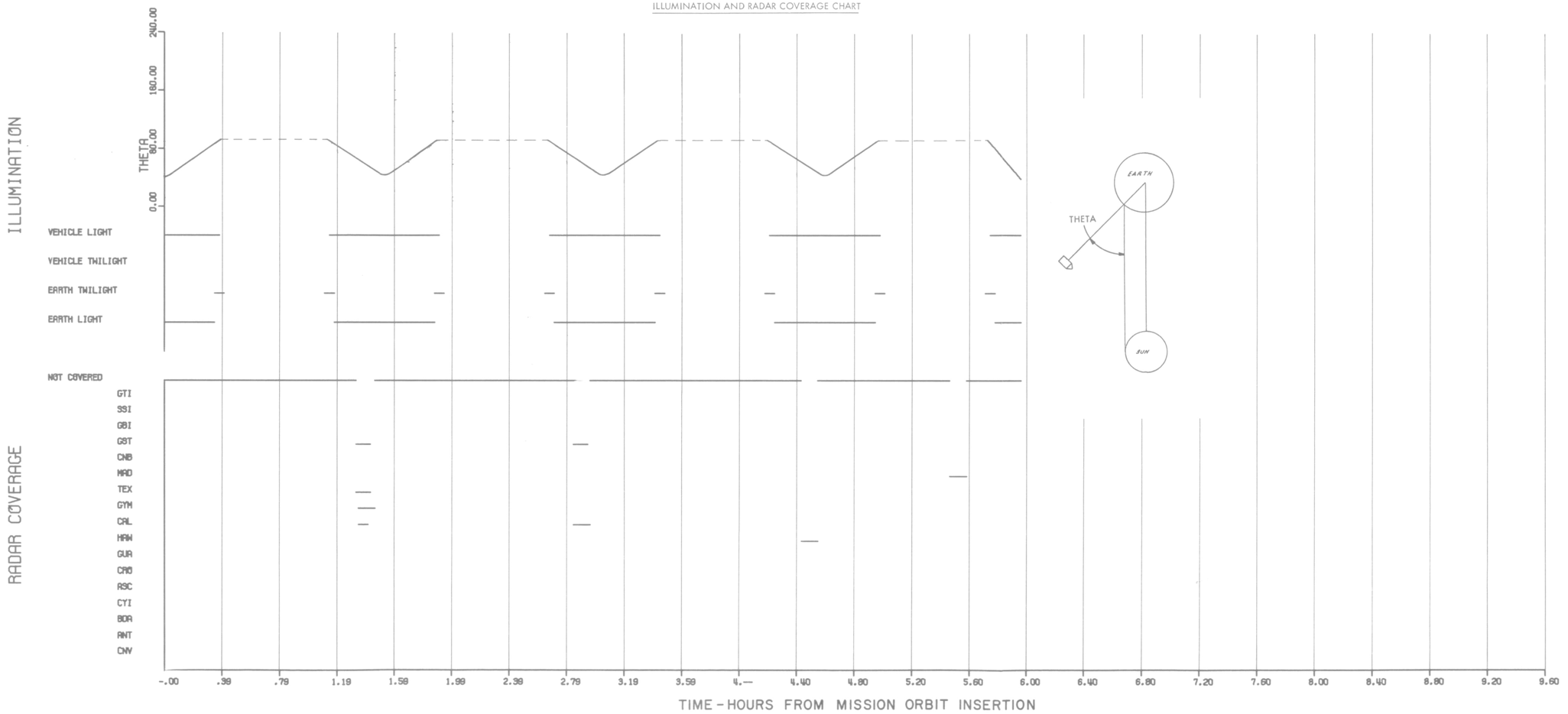

Figure 4-11. Illumination and Radar Coverage

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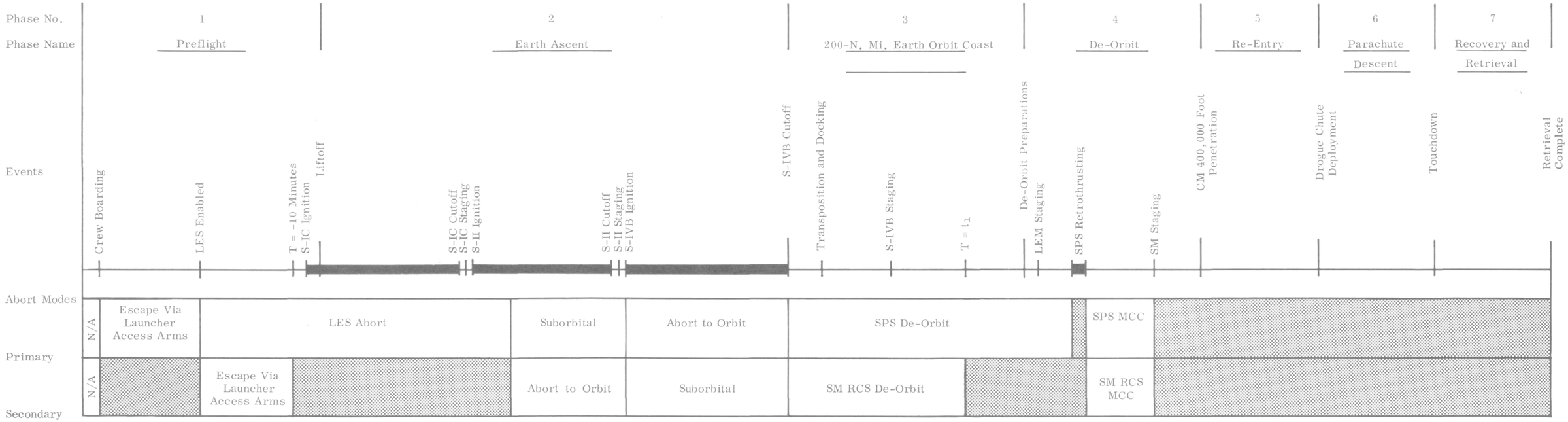


Figure 4-12. Mission Phase Versus Abort Mode

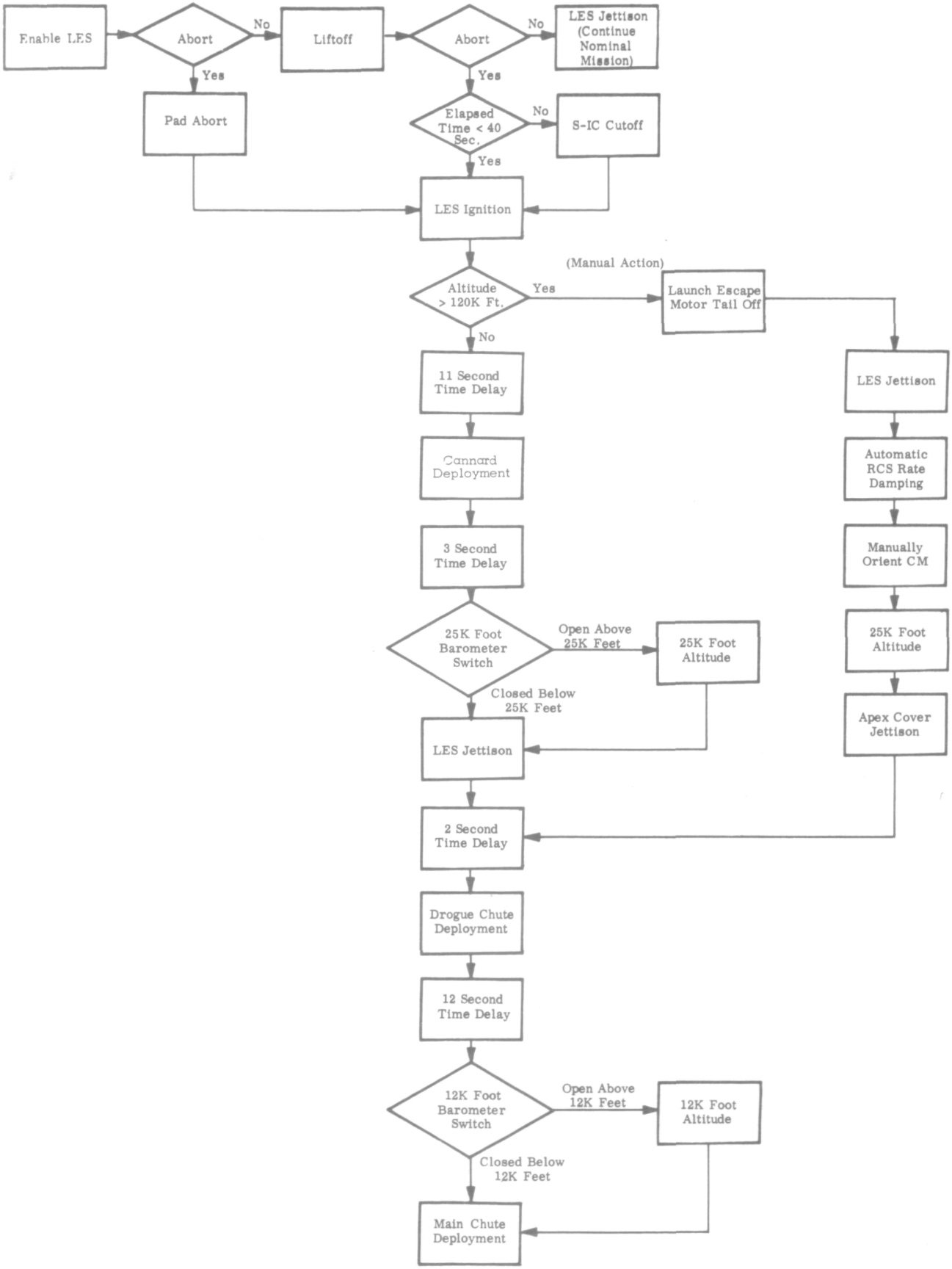


Figure 4-13. LES Abort

At abort initiation, launch vehicle engine shutdown is commanded (except during the 40-second range safety lockout). The launch escape and pitch control motors are then fired simultaneously. The pitch motor burns for approximately 0.5 seconds to provide lateral separation from the booster, while the escape motor thrusts for about 3.5 seconds to provide longitudinal separation.

At 11 seconds after abort, the canards are deployed. At 14 seconds after abort, if a 25,000-foot altitude barometric switch (closed during prelaunch) is still closed, the tower jettison motor separates the LES from the CM, including the CM apex cover. If the barometric switch is open, LES jettison is delayed until reaching 25,000 feet altitude on descent, when the barometric switch is closed again. Drogue parachute deployment occurs two seconds after LES jettison. Main parachute deployment is controlled by a timer and barometric switch. At 12 seconds after drogue deployment, if a 12,000-foot barometric switch is still closed, the drogue chutes are disconnected and the main chutes are deployed. If the switch is open, the drogue chutes are retained until reaching 12,000 feet on descent, when the switch is closed again.

For aborts above approximately 120,000 feet, the canard subsystem is ineffective and the pilot must manually orient the CM in a heat-shield-forward attitude. To do so, the LES must be manually jettisoned by the pilot after launch escape motor thrust tail-off (about 8 seconds). The CM apex cover is left on. Automatic RCS rate damping begins at tower jettison. Upon reaching 25,000 feet on descent, the apex cover is jettisoned upon closing of the barometric switch. Two seconds later the drogue parachutes are deployed. At 12,000 feet another barometric switch closes to disconnect the drogue chutes and deploy the main chutes.

4.2.6.3 Suborbital Aborts

4.2.6.3.1 Ground Rules

A suborbital abort can be initiated anytime between LES jettison and earth orbit insertion. A suborbital abort is the primary abort mode from LES jettison (during the S-II thrusting period) to the ignition of the Saturn S-IVB stage. A suborbital abort is the secondary abort mode from S-IVB ignition to the completion of the earth orbit insertion maneuver (to S-IVB cutoff). In both abort modes, the CM lands in a planned recovery area. Launch vehicle shutdown must precede initiation of spacecraft abort action.

The CSM abort separation procedure utilizes both a SM RCS and SPS burn of fixed initiation time, ΔV, and attitude. The SM must be separated from the CM at some fixed increment of time after SM engine shutdown following completion of the suborbital abort maneuvers.

Suborbital aborts before S-IVB ignition (primary abort mode) require that the CM fly a maximum lift trajectory during entry to avoid excessive decelerations. Suborbital aborts after S-IVB ignition (secondary abort mode) require SPS thrust to obtain the range of the landing site. A constant bank angle of approximately 60 degrees will be flown by the CM during entry.

4.2.6.3.2 Mode Description

A detailed sequence of events during a suborbital abort is shown in Figure 4-14. Launch engine shutdown must precede initiation of spacecraft abort action. Following booster shutdown, the SM RCS is used to provide approximately 45 inches of separation between the CSM and the launch vehicle to clear the SPS engine nozzle of the spacecraft/ LEM adapter (SLA). The SPS engine nozzle must be clear of the SLA as soon as possible to damp out any existing pitch and yaw rates that could exist at abort as a result of a launch vehicle control system failure. After the SM RCS separation and SPS damping maneuvers are completed, a turnaround to the required thrust attitude is achieved by pulsing the SM RCS.

Depending on the range control required, the SPS can be burned for variable lengths of time in either a posigrade or retrograde attitude. The time delay to thrust initiation can also be varied. After the CM is separated from the SM, the CM is capable of re-entering ballistically or with variable lift.

SPS range control thrusting is not used for aborts initiated during the period between LES jettison and S-IVB ignition. The CM, flying a maximum lift trajectory, can land within the confines of the planned recovery area. The procedure for the primary abort mode is to use the SM RCS to establish the correct CM/SM separation attitude after the SM RCS launch vehicle separation and SPS damping maneuvers are completed. The SM RCS must provide the CM/SM separation velocity as the CM has no translatory thrust capability. Following separation, the CM RCS is used to orient CM for a maximum lift entry.

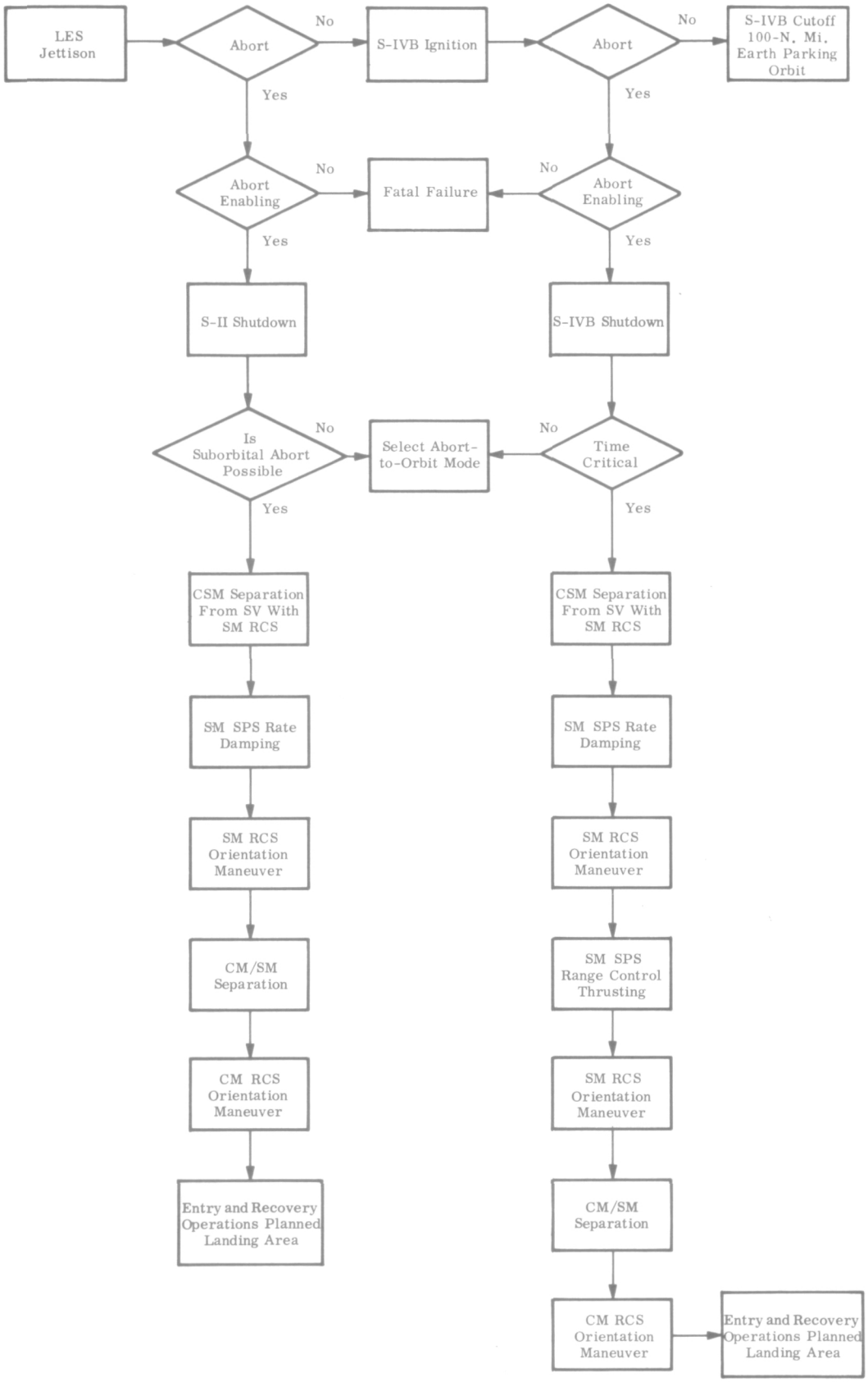


Figure 4-14. Suborbital Abort

During the period from S-IVB ignition to S-IVB cutoff (earth orbit insertion), the suborbital abort is a secondary abort mode. SPS range control thrusting will be utilized during this period to ensure a CM landing in the planned recovery area. Following SPS range control thrusting, the SM RCS is used to establish the correct CM/SM separation attitude, and then to provide the separation velocity. Following separation, the CM/RCS is used to orient the CM for a constant bank angle entry utilizing approximately half-lift capability.

4.2.6.4 Aborts to Orbit

4.2.6.4.1 General

The abort-to-orbit can be executed any time from S-IVB ignition to completion of the earth parking orbit insertion maneuver. All time-critical aborts during this period require the suborbital abort mode.

4.2.6.4.2 Ground Rules

An abort-to-orbit is the primary abort mode from the ignition of the S-IVB stage to the completion of the earth orbit insertion maneuver, i.e., to S-IVB cutoff. An abort-to-orbit is the secondary abort mode from LES staging during S-II thrusting to S-IVB ignition. Launch vehicle shutdown must precede initiation of spacecraft abort action. All time critical aborts will be suborbital.

The S-IVB or SM propulsion and guidance systems can be used for the abort-to-orbit mode. For those cases where SPS thrusting is required, the CSM abort separation procedure will utilize both a SM RCS and SPS burn of fixed initiation time, ΔV, and attitude.

Aborts-to-orbit after S-IVB ignition (primary abort mode) will utilize the SPS. Aborts-to-orbit prior to S-IVB ignition (secondary abort mode) are dependent on the following time constraints:

a. Aborts initiated prior to T + 140 seconds must utilize the S-IVB propulsion plus the SM propulsion.
b. Aborts initiated after T + 410 seconds but prior to T + 490 seconds must utilize the S-IVB propulsion.
c. Aborts initiated after T + 490 seconds must utilize the SPS.

The abort-to-orbit mode requires that a fixed amount of SM propellant be reserved for deorbit capability.

4.2.6.4.3 Mode Description

The advisability of aborting to orbit, once the capability exists, depends on the malfunction. If crew safety is not diminished, one advantage of aborting to orbit is the greater flexibility in landing site selection. A detailed sequence of events in an abort-to-orbit is shown in Figure 4-15. The S-IVB or SPS provides the necessary thrust to place the spacecraft into a 200-n.mi. circular earth orbit; however, the vehicle must be selected as dictated by the time of abort, the capability of each vehicle, and the propellant that must be reserved for de-orbit.

The SM SPS can supply the necessary ΔV to place the CSM in the 200-n.mi. orbit from an abort occurring about 490 seconds after liftoff. (An SPS abort-to-abort from an abort occurring at T + 490 seconds is accomplished by burning the SPS for approximately ( ) seconds.) The S-IVB can supply the necessary ΔV to place the CSM in the 200-n.mi. orbit from an abort occurring after about 410 seconds from liftoff. (An S-IVB abort-to-orbit from an abort occurring at T + 410 seconds is accomplished by burning the S-IVB for approximately ( ) seconds).

Both the S-IVB and SPS must be used for an abort-to-orbit for an abort occurring during the period from LES jettison to T + 410 seconds. The abort procedure requires thrusting with the S-IVB until burnout followed by an SPS burn to complete the orbit insertion maneuver.

Although the abort-to-orbit mode previously described indicates that the abort would be to the nominal 200-n.mi. orbital altitude, the abort could be to any altitude between 100 and 200 n.mi. with the lower limit determined by the orbital lifetime. An abort to an orbital altitude less than the nominal 200 n.mi. would permit the abort-to-orbit mode to be utilized at an earlier mission time, or permit an abort requiring less ΔV; because of a lower orbit, the time required to deorbit to an adequate landing area would be reduced.

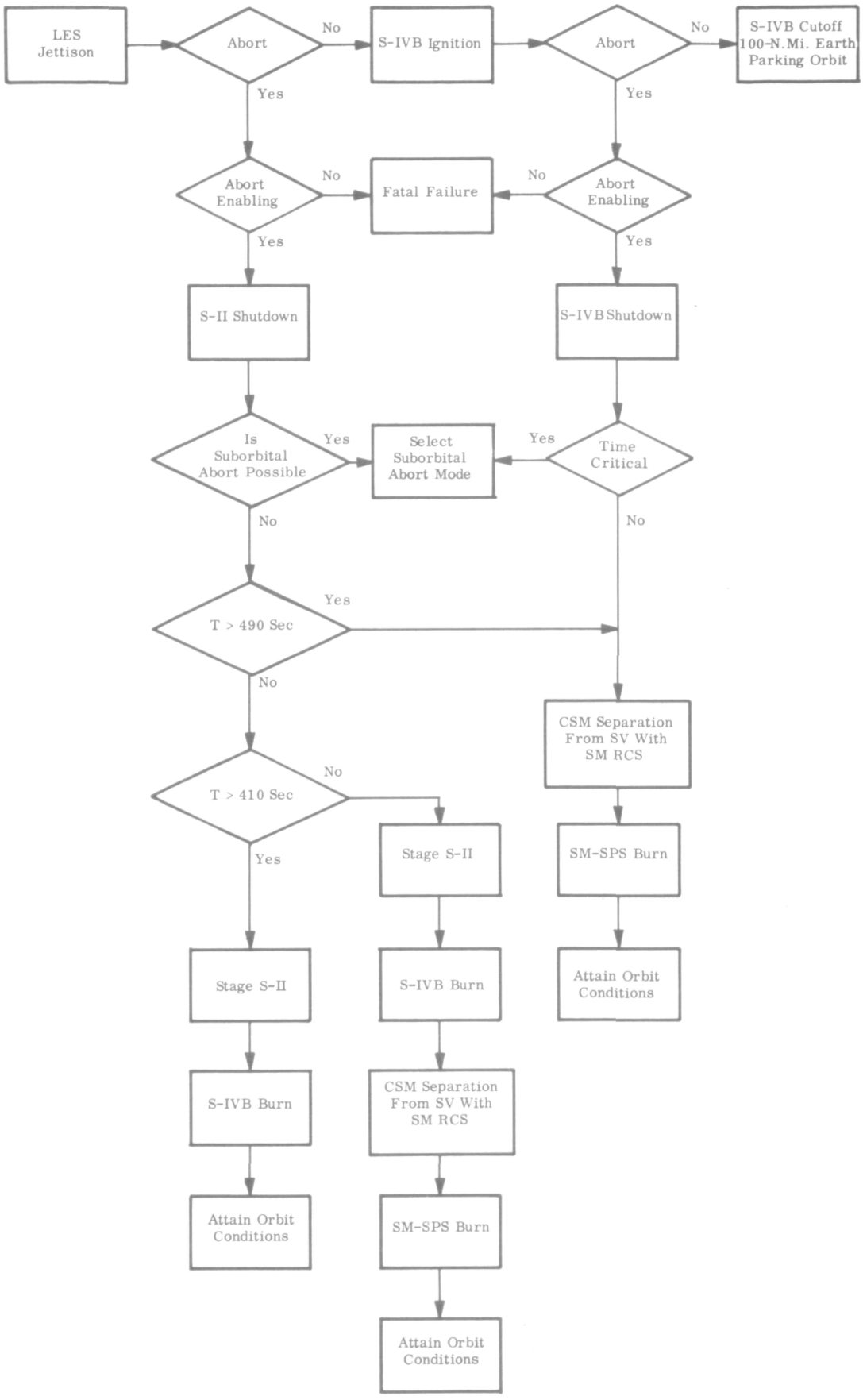


Figure 4-15. Abort to Orbit

4.2.6.5 De-orbit Aborts

4.2.6.5.1 General

The only abort possible from S-IVB cutoff at the completion of the earth orbit insertion maneuver to the ignition of the SPS for the nominal de-orbit maneuver is an early de-orbit maneuver. The primary abort mode utilizes the SPS to accomplish the de-orbit maneuver. The secondary abort mode utilizes the SM RCS to accomplish the de-orbit maneuver, but the secondary abort mode will not be available after a mission time of t_1.

4.2.6.5.2 Ground Rules

The spacecraft shall consist of only CSM at the time of initiation of de-orbit thrusting. The de-orbit maneuver will utilize a single propulsion burn at a fixed retrograde attitude and for a specified ΔV. The CM/SM separation procedure will utilize a SM RCS translation and roll maneuver.

When time permits, the de-orbit maneuver will be initiated at a time to permit entry to one of the planned recovery areas, preferably to the designated primary recovery area.

4.2.6.5.3 Mode Description

A detailed sequence of events during a de-orbit abort is shown in Figure 4-16. The primary de-orbit abort mode utilizes the SPS for the de-orbit maneuver. The secondary de-orbit abort mode utilizes the SM RCS to provide the de-orbit propulsion; however, the secondary abort mode is only available until $T = t_1$, the mission time at which the available SM RCS propellant becomes less than that required for the de-orbit maneuver.

Time critical aborts require immediate initiation of the de-orbit abort procedure with minimum consideration given to the resulting landing area. In non-time critical situations, the de-orbit maneuver can be delayed, if necessary, in order to return to an adequate landing area.

If abort action is required prior to S-IVB staging following transposition and docking, the abort procedure is to stage the S-IVB-IN-SLA-LEM, use the SM RCS to properly orient the CSM, and de-orbit by an SPS retro burn.

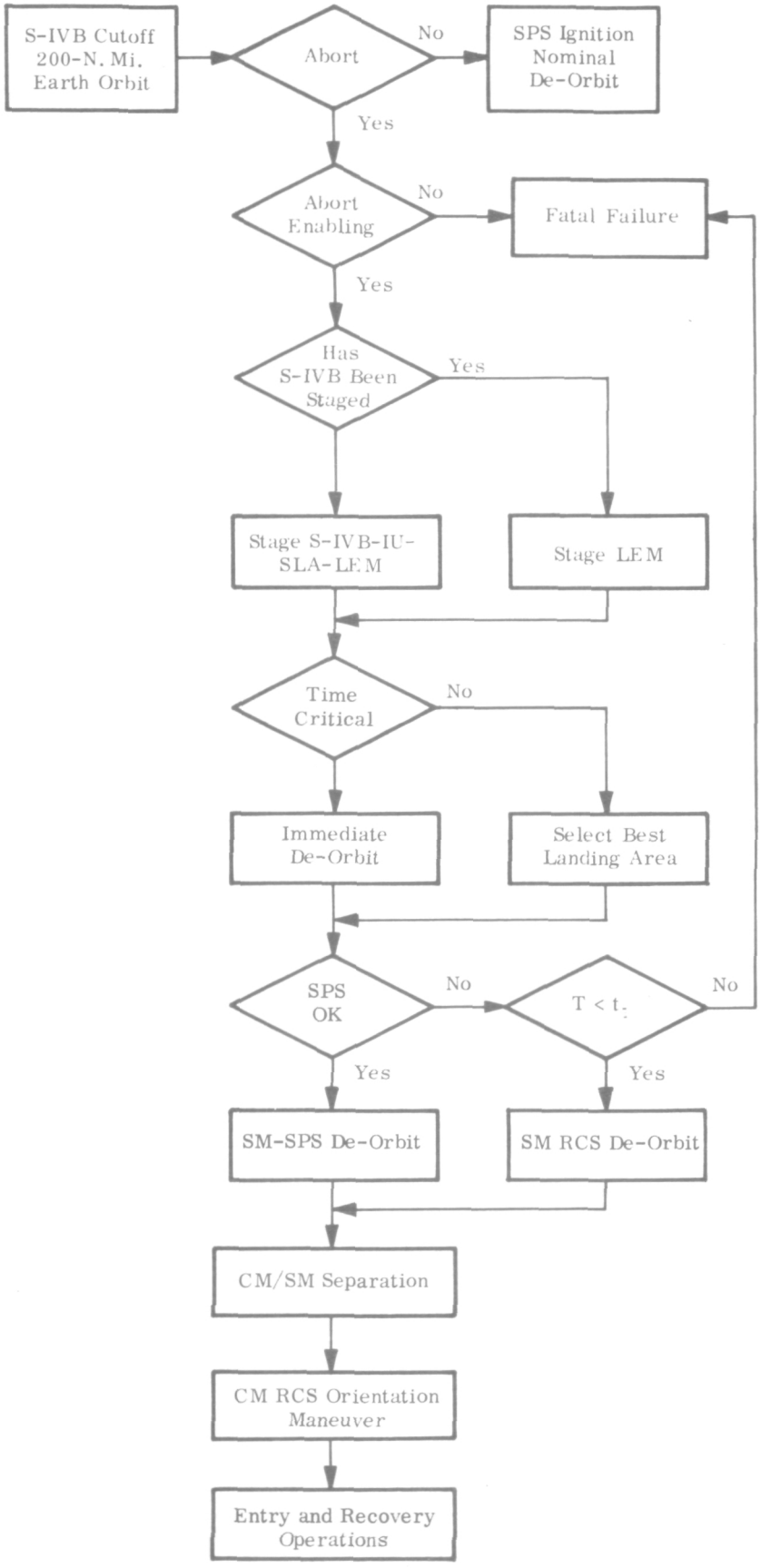


Figure 4-16. De-orbit Abort

If abort action is required after S-IVB staging following transposition and docking, the abort procedure is to stage the LEM, use the SM RCS to properly orient the CSM, and de-orbit by an SPS retro burn.

SECTION 5

MISSION 5

SATURN V TO SATURN IB RENDEZVOUS

5.1 MISSION DESCRIPTION

5.1.1 MISSION SUMMARY

5.1.1.1 Introduction

The Saturn V to Saturn IB Rendezvous mission will continue astronomical studies, gather data on living organisms in a space environment, conduct other experiments in the sciences and in space structures technology, and study spacecraft personnel and cargo transfer operations (see 1).

Mission 5 will consist of two flights, each manned by a crew of three. A basic Apollo Block II CSM, a modified LEM, a Launch Escape System (LES), and a spacecraft LEM adapter will comprise each spacecraft. The first flight, Vehicle A, will utilize the Saturn V launch vehicle; Vehicle B will utilize the Saturn IB (see 19).

Vehicle A will be placed in a nominal 200-n.mi. circular earth orbit at an inclination of 29.5 degrees. Nominally forty-four days later, Spacecraft B will be launched. The two spacecraft will rendezvous and dock and transfer the designated personnel and equipment. Spacecraft A will de-orbit and re-enter the earth atmosphere for recovery. Spacecraft B will remain in orbit, conducting its scheduled experiments for 45 days, then it will de-orbit, re-enter and be recovered (see 19).

5.1.1.2 Configuration

The launch vehicle used for the first launch of this mission will be the Saturn V, which consists of three propulsive stages (S-IC, S-II, and S-IVB) and a nonpropulsive module, the Instrument Unit (IU) (see 101-6).

The launch vehicle used for the second launch of this mission will be the Saturn IB, which consists of two propulsive stages (S-IB and S-IVB) and an IU (see 101-6).

The spacecraft used for each of the two flights comprising this mission will be a Block II CSM, a modified LEM, an LES, an SLA, and added expendables and subsystems to permit operations for 45 days (see 101-2).

5.1.1.3 Experiments

5.1.1.3.1 First Flight

Twenty types of experiments in astronomy and bioscience are planned for this flight (see 35-12). A substantial portion of the total experiment weight is occupied by the reflecting telescope (see 33).

The total weight of the experiment equipment is approximately 8900 pounds. It occupies 218 cubic feet and requires an average power of 680 watts, with peaks to 1350 watts (see 33).

5.1.1.3.2 Second Flight

There are six types of experiments planned for this flight (see 53-15), in conjunction with the 20 of the first flight. Experiments performed during Flight B are in astronomy, space logistics, and bioscience.

The six experiments utilize equipment that weighs approximately 1300 pounds and occupy 77 cubic feet (see 35).

5.1.1.4 Operations

Spacecraft A will be launched from Cape Kennedy Launch Complex 39B on an 83-degree launch azimuth and placed directly into the 200-n.mi. mission orbit by the Saturn V launch vehicle.

Transposition and docking is then accomplished, and the S-IVB, the IU, and the SLA are staged.

After the post-insertion tasks are completed, the interlock and LEM are pressurized and manned. Upon completion of the required LEM subsystem checks, the mission's prime experiments can be initiated.

The astronauts begin a work/rest cycle after the initial experiment preparations are completed. The crewmen perform, adjust, and calibrate any experiments or associated apparatus, perform station-keeping duties, and accomplish their normal spacecraft functions until several orbits prior to the launch of Spacecraft B, during the forty-fourth mission day. Launch of the second vehicle can be scheduled as early as the thirty-fifth day to allow for launch-date slippage. Mission time would advance accordingly.

During the forty-fourth mission day, Spacecraft B is launched from Cape Kennedy Launch Complex 37B and placed in a nominal 80-n.mi. circular earth-parking orbit by the two-stage Saturn IB launch vehicle. The launch azimuth and the inclination of the parking orbit will be based on mission requirements, but the parking orbit will be between 28.5 and 31.5 degrees, based on a plane change capability of one degree. This one-degree plane change capability is critically dependent on availability of fuel for orbital rendezvous fuel (see 50).

Vehicle B accomplished transposition and docking during the parking orbit. Shortly after a hard dock is made between CSM B and LEM B, the B vehicle's S-IVB, including the IU and SLA, is staged.

After three hours in parking orbit, the SPS inserts Spacecraft B into a co-planar Hohmann transfer orbit with an apogee of 200 n.mi. Spacecraft A and B track each other during the Spacecraft B ascent coast. All maneuvers prior to rendezvous are performed by Vehicle A. Depending on the relative phasing in orbit of the two vehicles, Vehicle A will be inserted into an elliptic orbit, having an apogee greater than 200 n.mi. to allow B to catch up, or into an elliptic orbit having a perigee less than 200 n.mi. to catch up with B. All information necessary for the computation of these elliptic orbits is available at the time that Vehicle B is launched. The maneuver may be performed at any later time in orbit.

Upon completion of the terminal rendezvous maneuvers, the two spacecraft are within a previously specified distance of each other. At this time LEM A separates from CSM A and accomplishes a hard dock with LEM B, which remains docked to CSM B. Following completion of LEM A docking to spacecraft B, CSM A accomplishes a hard dock to LEM A. At this point the two spacecraft are in a LEM-to-LEM docked configuration. This configuration is maintained until the planned crew, supply, and materiel transfer is complete.

Following transfer operations, preparations are made for CSM A separation. The CSM A crew effects a separation from the CSM B, LEM A, and LEM B, which continue in the 200-n.mi. mission orbit for the following 45 days. Then CSM A de-orbits, re-enters, and is recovered.

5.1.2 MISSION OBJECTIVES

5.1.2.1 Experiment Objectives

The experiment objectives of Mission 5 include studies in space structures technology, in crew and cargo transfer operations, and studies begun on previous missions in astronomy, bioscience, and the physical sciences.

The following Vehicle A experiments will be conducted during the first flight of Mission 5 (see 35-12):

a. 0100 Biomedical (excluding 0101 and 0102).
b. 0200 Behavioral.
c. 0501 Radiation Environment Monitoring.
d. 0601 Micrometeoroid Collection.
e. 0602A Kinetics and Dynamics of Vapor/Gas Bubbles.
f. 0602B Liquid Drop Dynamics Studies.
g. 0603A Pool Boiling in Long-Term Zero-g.
h. 0603B Nucleate Condensation of Fluids in Zero-g.
i. 0604 Density Gradient of a Fluid at Near-Critical State.
j. 0605 Crystallization Studies.
k. 0606 Cosmic Ray Emission.
l. 0701 Emission Line Radiometry.
m. 0702 Intermediate Size Reflecting Telescope.
n. 0703 Manned Coronagraph.
o. 0704 Gamma and X-ray Telescope.
p. 0705A Effects of Space Environments on Structures.
q. 1302 Deployment of RF Reflective Structures.
r. 1401 Personal Hygiene and Fuel Technology.
s. 1504 Development of Personnel and Cargo Transfer Operations.

All experiments begun on the first flight of the mission will be continued during the second flight. In addition, the following experiments will be conducted (see 35-15).

a. 0401 Microbial Genetics and Growth.
b. 0402 Development Biology.

c. 0403 Mammalian Physiology.

d. 0404 Limb Regeneration and Wound Healing.

e. 0405 Effect of Drugs on Mammalian Behavior.

f. 0406 Soft Capture, Enumeration, and Identification of Space-borne Microorganisms.

5.1.2.2 Flight Systems Objectives

5.1.2.2.1 Launch Vehicle - First Launch

A Saturn V launch vehicle (see 9-1, 9-10) will place the first of the two spacecraft payloads into the mission orbit of 200 n.mi. at a 29.5-degree inclination by direct ascent from launch, utilizing the first two Saturn V stages.

5.1.2.2.2 Launch Vehicle - Second Launch

A Saturn IB launch vehicle (see 9-1, 9-10) will place the second of the two spacecraft payloads associated with this mission type into an 80-n.mi. parking orbit that will permit the CSM to transpose and dock. The SPS will effect a Hohmann transfer from the appropriate point in the parking orbit to the 200-n.mi. mission orbit of Vehicle A. Rendezvous maneuvers are performed normally by Vehicle A. Thirty-five to 45 days can separate the launch dates of the two flights although a forty-four day separation is used for mission planning.

5.1.2.2.3 Spacecraft

The spacecraft for each of the two flights will (1) carry the mission experiments and crew to the specified mission orbit; (2) effect the rendezvous to yield the second flight spacecraft configuration of a CSM and two LEM's; (3) permit the transfer of crew personnel, supplies, and materials; (4) return the crew and data samples; and (5) provide operational support for the crew and experiments for 45 days.

5.1.2.2.4 Crew

The crew of each vehicle will consist of three astronauts (see 9-4) who are expected to participate in the performance of experiments beyond merely checking out and operating equipment. They will help ensure the scientific and technical validity of the experiments and provide partial evaluations of results (see 13).

One crew member will remain aloft for the full mission duration of both flights. The technical specialty of each crew member is unknown; therefore, the determination of which crew members will be exchanged during the rendezvous operations can not be defined at this time.

5.1.2.3 Ground Support Objectives

5.1.2.3.1 Mission Command Control

The mission command responsibility will remain the same as for the basic Apollo mission, with provision for handling two inflight vehicles simultaneously, and the ability to pass the mission command to the onboard crew commander for all but the critical mission phases (see 15).

5.1.2.3.2 Launch Support and Control

The Launch Control Center and normal complement of equipment at Pads 39B and 37B (see 7) will be utilized for both flights comprising this mission. Modifications to launch facilities and checkout operations will be minimized (see Figure 9-13). Assurance that the Saturn IB launch will occur 35 to 45 days following the Saturn V launch is necessary for mission success.

5.1.2.3.3 Tracking and Communications

All MSFN remote sites supporting this mission will be remoted directly to MCC-Houston. Sufficient data will be routed in real-time to preclude the necessity of remote site controllers (see 15-11). Tracking and communications are required continuously throughout the Saturn V and Saturn IB powered flight (see 15-66). Provisions to communicate and track both spacecraft, simultaneously when in near proximity of each other and working with a single MSFN site, requires the consideration of remote site frequency assignments and system coverage.

5.1.2.3.4 Recovery

The mission recovery operations will be similar to the basic Apollo system, with the Pacific Recovery Area remaining the primary recovery area (see 24). The potential use of recovery/rescue aircraft will be considered in lieu of continuous assignment of ships for world-wide abort coverage (see 9-70). When both vehicles are in flight, the ability to effect recovery of both vehicles during a one-to-three day time period must be considered.

5.1.2.3.5 Post Flight

The objectives for post-flight operations, involving such activities as data reduction, equipment performance review, and ground support performance reviews, were not considered in AAM program documentation.

5.1.3 MISSION GROUND RULES

5.1.3.1 Earth Launch Phase

The Merritt Island Areas (MILA) launch pad assignment for the launch of Flight 523 is Pad 39B; the assignment for Flight 229 is Pad 34B (see 29-1).

The launch azimuth for Flight A will be 83 degrees. Range safety launch azimuth limits will be 72 to 108 degrees, based on the earth-fixed heading of the launch vehicle at the beginning of the pitch program. Operational launch azimuth limits will be limited to a range of 26 degrees (see 29-1). As indicated by Table 1, launch azimuths for Flight B are variable depending on the launch time relative to some optimum.

Continuous tracking is required from launch to three minutes after initial orbit insertion (see 91).

5.1.3.2 Earth Parking Orbit Phase

The nominal earth parking orbit for Flight B (Saturn IB) will be a near-circular orbit of an 80-n.mi. altitude (see 24-4).

Transfer orbit injection for Flight B will occur not later than three orbits after injection (see 47-2). Injection will occur at the earliest opportunity after spacecraft systems have been checked out and the orbit ephemeris has been determined (see 29-2). For initial studies the third orbit was selected for the nominal injection orbit (see 29-2).

MSFN will be the primary source for trajectory determination (see 26-5). A minimum of two station contacts is required during the first parking orbit, one station contact during all subsequent parking orbits (see 14-33).

The astronaut will have control of the S-IVB reaction jets for attitude control from earth orbit insertion to separation of the CSM from the S-IVB/LEM (see 24-2).

5.1.3.3 Transfer Trajectory Phase

The transfer trajectory for Flight B will be a Hohmann ellipse (see 19).

There will be no natural lighting constraints on transposition and docking (see 26-3) except for the illumination of the docking tunnel. Ground rules on communications requirements during transposition and docking can not be written at this time.

One four-minute station contact with the ground network is required within 90 minutes before burn (see 91); one four-minute contact is required not later than seven minutes after burn (see 91).

5.1.3.4 Mission Orbit Phase

The nominal mission orbit will be a near-circular orbit at a 200-n.mi. altitude (see 19). Orbit inclination will be 29.5 degrees (see 50-7).

Flight mission is under control of the flight director who, under normal circumstances, will delegate control of the scientific program after the spacecraft has achieved its designated orbit path (14-55).

One four-minute station contact with MSFN is required per orbit. The time between two successive contacts will not exceed two hours (see 91).

The spacecraft will permit extravehicular transfer of the crew between the CM and the LEM in either direction, in both the docked and undocked condition (see 24-3). Docking maneuvers will be controlled manually (see 24-2). In all earth-orbital missions, the SPS will be used for de-orbiting maneuvers (see 14-63).

At least one astronaut in the CM will be wearing a spacesuit at all times during flight. This may be with a faceplate either open or closed (see 24-2). No more than two astronauts will be off duty at any one time during the mission (see 76-5).

One crew member will be exchanged from Spacecraft A to Spacecraft B (see 47).

The following are proposed as candidate rendezvous ground rules based on the results of the trajectory analysis:

a. Spacecraft A will actively perform all maneuvers required for rendezvous.

b. Spacecraft B will fly a nominal mission and remain passive during the rendezvous maneuvers.
c. The primary command, control, and computation of coarse rendezvous maneuvers will be the responsibility of MCC.
d. The primary responsibility for terminal rendezvous maneuvers will rest with commander of Spacecraft A.
e. A prerendezvous maneuver will be completed by Spacecraft A prior to the lift off of spacecraft B to correct the phasing position to correspond to a zero-degree plane change launch.
f. The launch window of spacecraft B is 180 minutes.

The LEM/LAB and CSM will dock on the supply spacecraft separately and in the order LEM/LAB-CSM (see 38-8). The coast duration in the docked configuration will be not greater than 24 hours (see 48).

5.1.3.5 Re-entry Phase

The following are the re-entry trajectory characteristics of Mission 5.

a. Vacuum perigee - zero n.mi. (see 49-2).
b. Re-entry range - 1500 to 3500 n.mi.
c. The nominal equatorial inclination of the inertial velocity vector during re-entry will be 29.5 degrees. The re-entry trajectory will be posigrade.
d. Cross range maneuvering - the nominal entry trajectory will not require cross-range maneuvering capability (see 24-5).

One 4-minute preretro contact is required with MSFN within 90 minutes prior to retro; one 4-minute post-retro contact is required not later than 7 minutes after retro (see 91).

The primary recovery area will be the Pacific Recovery Area (see 29-9).

5.1.4 WEIGHTS SUMMARY

Table 5-1 and 5-2 summarize the spacecraft weights for Mission 5, Flights 523 (Saturn V), and Flight 229 (Saturn IB). Each table is broken into three sections: Command Module, Service Module, and LEM/LAB. For each module the major categories are listed and referenced.

The Command Modules are NAA Block II Modification D configuration (see 45); the Service Modules are Block II Modification D configurations (see 45), except for RCS

Table 5-1

Mission No. 5, Saturn V to Saturn IB Rendezvous
Flight 523 - Spacecraft Weights Summary (First Launch)

Module	Weight - lbs		Reference No. and Page
1. Command Module			
Empty Weight	9,034		(45) No page
Useful Weight	1,269		(45) No page
Total	[10,303]		
2. Service Module			
Empty Weight	8,921		(45) No page
Reaction Control Propellant[1]	2,457		
Experiments RCS Propellant		(720)	(31) 1.13.38
Excess RCS Propellant		(1,737)	
Cryogenic Expendables	653		(45) No page
Service Propulsion Fuel	41,000		
Retro[2]		(824)	Part II, Sect. 6
Plane Change		(12,700)	Para 5.2.5
Excess		(27,476)	
Total	[53,031]		
3. LEM/LAB			
Structure	2,707		(33) 1.13.36
SCS	43		(33) 1.13.36
NGS	126		(33) 1.13.36
Crew Provisions	314		(33) 1.13.36
RCS	4,616		(33) 1.13.36
(Propellant)		(3,360)	(33) 1.13.36
Controls and Displays	779		(33) 1.13.36
Environmental Control	1,333		(33) 1.13.37
Instrumentation	171		(33) 1.13.37
EPS	5,021		(33) 1.13.37
(Reactants and O_2)		(2,608)	(33) 1.13.37
Communications	58		(33) 1.13.37
Experiments[3]	8,948		(35) Para 10.5
Spares	60		(33) 1.13.37
Total	[24,176]		
4. Total Spacecraft Weight[4]	87,510		
Maximum Spacecraft Weight into Orbit	232,956[5]	110,000[6]	(111) No page
Margin	145,446	22,490	

1. The RCS tank capacity.
2. This figure includes a 10 percent ΔV reserve.
3. The experiments package weight represents a reassignment of Experiment 1302 and 1504 (weight 1,161 pounds) to Flight 523 from Flight 229.
4. Summation of square bracketed numbers.
5. No stack limit above S-IVB and IU stage.
6. Stack limit.

Table 5-2

Mission No. 5, Saturn V to Saturn IB Rendezvous
Flight 229 - Spacecraft Mission Orbit Injection Weights Summary

Module	Weights - lbs		Reference No. and Page
1. Command Module			
Empty Weight	9,034		(45) No page
Useful Weight	1,269		(45) No page
Total	[10,303]		
2. Service Module			
Empty	8,921		(45) No page
Reaction Control Propellant[1]	720		(33) 1.14.38
Experiments RCS Propellant		(720)	
Excess RCS Propellant		(0)	
Cryogenic Expendables	653		(45) No page
SPS Retrofuel[2]	824		Part II, Sect. 6
Total	[11,118]		
3. LEM/LAB			
Structure	2,017		(33) 1.14.38
Crew Provisions	296		(33) 1.14.38
RCS	1,470		(33) 1.14.38
(Propellant)		(970)	(33) 1.14.38
Controls and Displays	290		(33) 1.14.38
Environment Control	981		(33) 1.14.38
Instrumentation	151		(33) 1.14.38
EPS	4,704		(33) 1.14.39
(Reactants and O_2)		(2,465)	(33) 1.14.39
Communications	32		(33) 1.14.39
Experiments[3]	1,298		(35) Para 11.1
Spares	60		(33) Para 1.13.39
Total	[11,299]		
4. Total Spacecraft Weight[4]	32,720		
Maximum Spacecraft Weight into Orbit[5]	34,685		(49) Page 10
Margin	1,965		

1. SM RCS capacity 2,457 pounds.
2. This figure includes a 10 percent ΔV reserve.
3. This experiment package weight represents a reassignment of Experiment 1302 and 1504 (weight 1,161 pounds) to Flight 523 and Experiment 1306 (weight 540 pounds) to Flight 530, flight is not reported in this study.
4. Summation of all square bracketed numbers.
5. Maximum weight in orbit was increased by 1500 pounds over the value cited in 49-10 (see 130).

fuel loading, which is based on computations assuming impulse burns and an additional 10 percent ΔV margin included in the retro fuel budget.

The LEM/LAB weights are based on GAEC experiment assignments (see 31) as altered by Bellcomm (see 35) to meet margin requirements.

5.1.5 EXPERIMENT LOAD DESCRIPTION

The Mission 5 experiment load descriptions are shown in Tables 5-3 and 5-4. There are 20 types of experiments planned for Flight A of this mission (see 35-12). The total weight of the experiment equipment is 8948 pounds, of which about one-third is a telescope. There are six additional types of experiments planned for Flight B (see 35-15). The total weight of Vehicle B experiment equipment is about 1300 pounds.

Most of the experiments planned for this mission are described in Mission 2, paragraph 2.1.5. The experiments involving deployment of RF reflective structures is unique to this mission. For this experiment, ground based operational support entails radar observations at as many frequencies as possible after the structure is deployed. Deployment, outside the vehicle, will require ground-based evaluations of the passive RF return of ground signals.

Table 5-3

Experiment Load Summary

(35-12) (35-12) (35-12) (33-1-13.4)

Experiment Number	Experiment Category/Abbreviated Title	Total Wt (lbs)	Volume (ft^3)	Volume Reference	Power (watts)	
					Avg	Peak
0100	Biomedical (Excluding 0101 and 0102)	301	14.5	(33-1-13.6)	100	50
0200	Behavioral	20				
0501	Radiation Environment Monitoring	47	2.64	(33-1-13.6)	6.3	
0504	Micrometeoroid Collection	95	22.1	(33-1-13.6)	1.01	
0601	Capillary Studies	113	3.63	(33-1-13.6)	430	
0602A	Kinetics and Dynamics of Vapor/Gas Bubbles	41	0.5	(33-1-13.6)	100	
0602B	Liquid Drop Dynamic Studies	71	1.6	(33-1-13.6 & .7)	2.2	1350
0603A	Pool Boiling in Long-Term Zero-g	52	2.57	(33-1-13.7)	180	550
0603B	Nucleate Condensation of Fluids in Zero-g	21	1.25	(33-1-13.7)	90	500
0604	Density Gradient of a Fluid at Near-Critical State	42	3	(33-1-13.7)	200	
0605	Crystallization Studies	152	4.5	(33-1-13.7)	500	
0606	Cosmic Ray Emission	60	6.53	(33-1-13.7)	60	100
0701	Emission Line Radiometry	220	8.62	(33-1-13.8)	75	
0702	Intermediate Size Reflecting Telescope	3680	18.5	(33-1-13.8)	200	
0703	Manned Coronagraph	260	8.1	(33-1-13.8)	133	
0704	Gamma and X-ray Telescope	310	11.2	(33-1-13.8)	60	
0705A	Effects of Space Environments on Structures	492*	13.43	(33-1-13.8 & .9)	722	
1302	Deployment of RF Reflective Structures	159	10.75	(33-1-14.7)	750	
1401	Personnel Hygiene and Fuel Technology	189	54.83	(33-1-13.9)	100	400
1504	Development of Personnel and Cargo Transfer Operations	1002	29.65	(33-1-14.8)	825	
Totals		8948	217.91		680	790

*1621.4 - (see 33-1-13.8)

Table 5-4

Experiment Load Summary

(35-15) Experiment Number	(35-15) Experiment Category/ Abbreviated Title	(35-15) Total Wt (lbs)	(35-15) Volume (ft^3)	Volume Reference	(33-1-14A) Power (watts)	
					Avg	Peak
0401	Microbial Genetics and Growth	60	7.7	(33-1-14.5)	30	50
0402	Development Biology	175	21.5	(33-1-14.5)	343	
0403	Mammalian Physiology	236	16.1	(33-1-14.6)	30	
0404	Limb Regeneration and Wound Healing	78	4.67	(33-1-14.6)	80	
0405	Effect of Drugs on Mammalian Behavior	708	25.5	(33-1-14.7)	56	
0406	Soft Capture, Enumeration, and Identification of Spaceborne Microorganisms	41	2	(33-1-14.7)	110	25
Totals		1298	77.47			

Table 5-5

Mission 5, Saturn-V/Saturn-IB Rendezvous Mission Phases

Phase	Initiating Event	Event Time Hrs Min Sec	Concluding Event	Duration ΔT Hrs Min Sec
1.0 Preflight		T - 485 hrs	Liftoff (T - 0)	485 hrs
2.0 Vehicle A Earth Ascent	Liftoff (T - 0)	0	EOI-S-IVB Cutoff	1014.32 sec(1)
2.1 S-IC Thrusting	Liftoff (T - 0)	0	S-IC Staging	162.37 sec(1)
2.1.1 Coast		158.57 sec(1)		3.8 sec(1)
2.1.2 S-IC Staging		162.37 sec(1)		
2.2 S-II Thrusting	S-IC Staging	162.37 sec(1)	S-II Staging	371.82 sec(1)
2.2.1 LES Jettison		193.53 sec(1)		
2.2.2 Coast		534.19 sec(1)		4.8 sec(1)
2.2.3 S-II Staging		538.99 sec(1)		
2.3 S-IVB Thrusting	S-II Staging	538.99 sec(1)	EOI-S-IVB Cutoff	475.33 sec(1)
3.0 Vehicle A 200 N.Mi. Earth Orbit	EOI-S-IVB Cutoff	16 m, 54 s	Beginning of Rendezvous Preparations	43 days
3.1 Initial Coast	EOI-S-IVB Cutoff		CSM/LEM - S-IVB Separation	
3.2 Transposition and Docking	CSM/LEM - S-IVB Separation		Hard Dock	40 min(3)
3.3 Coast through S-IVB Staging	Hard Dock		S-IVB Staging	
3.4 Earth Orbit Operations	S-IVB Staging		Beginning of Rendezvous Preparations	

Table 5-5

Mission 4, Saturn-V/Saturn-IB Rendezvous Mission Phases (Cont.)

Phase	Initiating Event	Event Time Hrs Min Sec	Concluding Event	Duration ΔT Hrs Min Sec
4.0 Rendezvous Preparations	Beginning of Rendezvous Preparations		First Terminal Maneuver	
4.1 Vehicle B Preflight	Preflight	25 days	Liftoff of Vehicle B	450 hrs
4.2 Prerendezvous maneuvering of Vehicle A	Vehicle A maneuvering	43 days(5)	Liftoff of Vehicle B	1 day
4.3 Rendezvous maneuvering of Vehicle A	Liftoff of Vehicle B	44 days	First terminal maneuver	3 hrs, 55 m(6)
5.0 Vehicle B Earth Ascent	Vehicle B Liftoff	44 d, 0 s	Parking Orbit Insertion - S-IVB Cutoff	595.54 sec
5.1 S-IB Thrusting	Vehicle B Liftoff	44 d, 0 s	S-IB Staging	145.94 sec
5.1.1 Coast		44 d, 141.64 s(1)		4.3 sec
5.1.2 S-IB Staging		44 d, 145.94 s(1)		
5.2 S-IVB Thrusting	S-IB Staging	44 d, 145.94 s(1)	Parking Orbit Insertion - S-IVB Cutoff	449.60 sec
5.2.1 LES Jettison		44 d, 165.94 s(1)		

Table 5-5

Mission 5, Saturn-V/Saturn-IB Rendezvous Mission Phases (Cont.)

Phase	Initiating Event	Event Time Hrs Min Sec	Concluding Event	Duration ΔT Hrs Min Sec
6.0 Vehicle B Earth Parking Orbit	Parking Orbit Insertion - S-IVB Cutoff	44 d, 9 m, 55.54 s[1]	TOI-SPS Ignition	3 hrs, 0 m[2]
6.1 Initial Coast	Parking Orbit Insertion - S-IVB Cutoff		CSM/LEM - S-IVB Separation	
6.2 Transposition and Docking	CSM/LEM - S-IVB Separation		Hard Dock	40 min[3]
6.3 S-IVB/IU/SLA Staging	Hard Dock		TOI-SPS Ignition	
7.0 Vehicle B Transfer Orbit	TOI-SPS Ignition	44 d, 3 h, 10 m	First Terminal Rendezvous Maneuver Ignition	45 min 22.7 s
7.1 TOI Thrusting	TOI-SPS Ignition		TOI-SPS Cutoff	11.5 sec
7.2 Transfer Orbit Coast	TOI-SPS Cutoff			45 min
7.3 EOI Thrusting	EOI-SPS Ignition	44 d, 3 h, 55 m	EOI-SPS Cutoff	11.24 sec

Table 5-5

Mission 5, Saturn-V/Saturn-IB Rendezvous Mission Phases (Cont.)

Phase	Initiating Event	Event Time Hrs Min Sec	Concluding Event	Duration ΔT Hrs Min Sec
8.0 Rendezvous of Vehicle A and Vehicle B	First Terminal Rendezvous Maneuver Ignition	44 d, 3 h, 55 m[6]	CSM A Hard Dock	1 hr, 27 m[7]
8.1 Terminal Rendezvous	First Terminal Rendezvous Maneuver Ignition		CSM/LEM A Separation	
8.1.1 First Terminal Rendezvous				
8.1.2 Second Terminal Rendezvous				
8.1.3 Third Terminal Rendezvous				
8.2 LEM A Docking Operations	CSM/LEM A Separation		LEM A - S/C B Hard Dock	
8.2.1 LEM A/CSM A Separation				
8.2.2 LEM A Begin Soft Dock with S/C B				
8.2.3 LEM A Hard Dock with S/C B				
8.3 CSM A Begin Soft Dock with S/C B LEM A				
8.4 CSM A Hard Dock with S/C A LEM A				

Table 5-5

Mission 5, Saturn-V/Saturn-IB Rendezvous Mission Phases (Cont.)

Phase	Initating Event	Event Time Hrs Min Sec	Concluding Event	Duration ΔT Hrs Min Sec
9.0 Docked Configuration Coast	CSM A Hard Dock	44 d, 5 h, 22 m	CSM A Separation	1 day [4]
9.1 Crew, Supply, and Material Transfer	CSM A Hard Dock		Beginning of CSM A Preparations for Separation	
9.2 CSM A Preparation for Separation	Beginning of CSM A Preparations for Separation		CSM A Separation	1 hr, 30 m
9.2.1 CSM A Separation				
10.0 CSM A De-Orbit	CSM A Separation	45 d, 5 hr, 22 m	Re-entry	27 min
10.1 SPS Retro Thrusting	CSM A Separation		SPS Retro Cutoff	10.5 sec
10.2 De-orbit Coast	SPS Retro Cutoff		SM Staging Ignition	17 min
10.3 SM Staging	SM Staging Initiation		Re-entry	10 min
11.0 Re-entry	Re-entry (400,000')	45 d, 5 h, 49 m	Drogue Chute Deployment	10 min [3]
12.0 Parachute Deployment	Drogue Chute Deployment	45 d, 5 h, 59 m	Touchdown	7 min [3]

Table 5-5

Mission 5, Saturn-V/Saturn-IB Rendezvous Mission Phases (Cont.)

Phase	Initiating Event	Event Time Hrs Min Sec	Concluding Event	Duration ΔT Hrs Min Sec
13.0 Recovery and Retrieval	Touchdown	45 d, 6 h, 06 m	Retrieval	
14.0 S/C B Plus LEM A Earth Orbit Coast	CSM A Separation	45 days	Beginning of De-orbit Preparations S/C B	44 days
15.0 CSM B De-orbit	Beginning of De-orbit Preparations S/C B	89 days	Re-entry	
15.1 LEM A and LEM B Staging	Beginning of De-orbit Preparations S/C B	89 days	LEM A and B Jettison	1 hr, 30 m
15.2 SPS Retro Thrusting	LEM A and B Jettison	89 d, 1 h, 30 m	SPS Retro Cutoff	10.5 sec
15.3 De-orbit Coast	SPS Retro Cutoff		SM Staging Initiation	17 min
15.4 SM Staging	SM Staging Initiation	89 d, 1 h, 47 m	Re-entry	10 min
16.0 Re-entry	(400,000') Re-entry	89 d, 1 h, 57 m	Drogue Chute Deployment	10 min(3)

Table 5-5

Mission 5, Saturn-V/Saturn-IB Rendezvous Mission Phases (Cont.)

Phase	Initiating Event	Event Time Hrs Min Sec	Concluding Event	Duration ΔT Hrs Min Sec
17.0 Parachute Deployment	Drogue Chute Deployment	89 d, 2 hr, 07 m	Touchdown	7 min(3)
18.0 Recovery and Retrieval	Touchdown	89 d, 2 hr, 14 m	Retrieval	

1. Reference 49, Tables 9 and 1.
2. Crew task requirements of 2 hours, 52 minutes included, as established by communications requirements.
3. Reference 26.
4. Reference 48 - Hard Dock Time of 1 day established to allow ample transfer time.
5. Prerendezvous maneuvering required to establish proper Vehicle A phasing for the nominal time of Vehicle B liftoff.
6. It is assumed for this time chart that any launch delay of Vehicle B from nominal liftoff can be compensated by the maneuvering of Vehicle A in the 3 hours 55 minutes.
7. Reference 24 - Crew tasks adapted.

5.2 OPERATIONS

5.2.1 MISSION PHASE DESCRIPTION

5.2.1.1 Introduction

Mission 5, for purposes of analysis and description, comprises 18 major phases. Table 5-5 shows each major phase, its initiating event, the initiating event time, the concluding event, and the time duration of each phase. The phase descriptions detail the operations conducted and the crew activities performed during each phase.

There are many parameters in this section that currently do not have assigned values. These parameters were included, based on comparable work in the AMPTF Design Reference Mission, in order to establish the basis for future development of the mission phase descriptions. A blank enclosed by parentheses will identify these parameters. Details of the functions during experiment activities are outlined in Appendix D Volume 3. Scheduling of experiments and flight crew operations are shown in paragraph 5.2.3.

The following is a detailed description* of Mission 5 by phases. The crew task descriptions also are based on Reference 25, the Apollo Design Reference Mission. Appendix C documents the crew activities prepared from the DRM adapted information.

5.2.1.2 Phase 1: Preflight

The preflight phase begins with the securing of the launch umbilical tower (LUT) to the launch pad and concludes with launch vehicle at T - 0.

The LUT is secured to the pad, the interfaces are checked, and all space vehicle-to-ground facilities are verified.

The space vehicle checkout begins with the GNS. Included in this subphase are launch vehicle propellant tanking tests, verification of all interface connections, spacecraft fueling operations, and a simulated spacecraft flight. All spacecraft systems are then shut down at T - 120 hours.

*This description is based on References 24, 26, and 29 and the summary charts on the AAM Earth Orbit Missions, Tables 5-1 and 5-2 of this volume.

The precountdown begins 485 hours before the scheduled launch time and continues until terminal countdown 10 hours prior to launch. During this period, the LEM and CSM receive cryogenic servicing, the GNS of both the CSM and LEM are checked out and activated, and the launch vehicle propellant is loaded. Other operations include the installation of cabin, crew, and water supplies and the final installment of LEM scientific equipment.

Ten hours prior to launch, the terminal countdown begins. The CSM is cleaned, sterilized, and closed out. The launch vehicle obtains its cryogenic load while tracking and communications networks are tested. At T -6 seconds, the S-IC engine ignition sequence is started and at T -0 the vehicle lifts off, ending the preflight phase.

5.2.1.3 Phase 2: Earth Ascent

The earth ascent phase (see Appendix C) begins with the Saturn V liftoff at T -0 and concludes with S-IVB cutoff after earth-parking orbit insertion.

After a six-second thrust buildup period from S-IC ignition, the vehicle is released and lifts off. The vehicle rises vertically (pitch = 90 degrees) for a 12-second period and then begins a programmed pitch maneuver that smoothly reduces the pitch angle to 23 degrees at the burnout point. At 30,000 feet, the dynamic pressure reaches a peak of 690 pounds per square foot. At S-IC burnout, at T +158 seconds, the vehicle velocity and approximate altitude are 8100 feet per second and 225,000 feet. The S-IC booster is staged prior to S-II ignition.

During ascent the commander monitors acceleration, attitude, attitude rates, abort lights, and pitch and roll rates. The navigator is responsible for monitoring the S-IC ignition sequence, the caution and warning indicators, and the environmental control system (ECS) panel. For the duration of the ascent phase, the engineer maintains radio communication with the control center and monitors the critical temperature flow and events displays. The commander reports the S-IC engine cutoff and staging sequence using the two-way voice communication link.

After S-IC staging, the vehicle coasts 3.8 seconds, and the S-II ignition occurs at T +162 seconds. The LES is jettisoned 31 seconds after S-II ignition. The vehicle executes a pitch maneuver that reduces the pitch angle to 5.95 degrees at S-II burnout.

The commander confirms the LES jettison, which occurs 31 seconds after S-II ignition. The S-II engines shut down approximately 9 minutes after launch on a signal from the propellant level sensors. The commander monitors this cutoff and staging sequence.

S-II burnout occurs at T +534.19 seconds, with a vehicle velocity and altitude of 17,454 fps, and 200 n.mi. At this point, the S-II and the CSM/LEM-S-IVB configuration is allowed to coast for 4.8 seconds before S-II staging and S-IVB ignition occurs.

The S-IVB is ignited at T +538.99 seconds. The vehicle performs a programmed pitch maneuver that terminates with S-IVB cutoff and vehicle insertion into a 200-n.mi. earth-parking orbit. S-IVB cutoff occurs at T +16 minutes, 54 seconds. The vehicle velocity at S-IVB cutoff is 25,231 feet per second.

5.2.1.4 Phase 3: Vehicle A Earth Orbit

The 200-n.mi. earth orbit phase (see Appendix C) for Vehicle A begins with the cutoff of the S-IVB and terminates with crew preparation for the rendezvous with Vehicle B.

After S-IVB engine cutoff, the commander and navigator of Vehicle A confirm orbit achievement. All three crewmen then adjust their couch restraints and set the controls and displays for earth-orbit operations. Using the inflight test system, the navigator and engineer perform a checkout of spacecraft systems, including guidance and control, environmental control, reaction control, and electrical power. The engineer also continues to maintain communications with the MCC.

The commander next stabilizes the spacecraft attitude so that the navigator can perform the IMU fine alignment and landmark navigational sightings. The S-IVB reaction control system is used to orient the space vehicle for landmark acquisition.

Once he has determined and verified the orbit ephemeris, the commander makes the decision for mission continuation in the normal mode.

The spacecraft is then oriented for transposition and docking. After the S-IVB stabilization control system (SCS) is placed in the attitude hold mode, the commander initiates the transposition maneuver by actuating the separation of the S-IVB from the CSM.

When the S-IVB-LEM/CSM separation is assured, the SLA is deployed to aid in the stabilization of the S-IVB-LEM and to expose the LEM. The CSM then translates 35 feet forward, performs a 180-degree pitch maneuver, and translates back to the S-IVB-LEM. The CSM performs docking maneuvers to align the CM-LEM interface and to engage the LEM drogue (X-axis docking tunnel). A maximum of 15 minutes is required from separation to the accomplishment of a soft dock. After soft docking has been achieved, the navigator secures the docking latches and electrical connections between the two modules and pressurizes the LEM.

A maximum of 25 minutes is required from soft dock to the securing of the CM latches after performing the hard dock.

Upon completion of this hard docking, the commander initiates the S-IVB separation from the CSM/LEM configuration, a sequence that is visually monitored by the engineer. The commander then translates the CSM/LEM from the jettisoned S-IVB stage. The engineer reports the maneuver completion to MCC.

A maximum of 5 minutes is required from hard docking to the completion of S-IVB/CSM-LEM separation.

Transposition and docking must be accomplished prior to a mission time of 6.5 hours, since the specified lifetime for the S-IVB-IU is 6.5 hours after liftoff.

While the engineer continues to record PCM telemetry and maintain ground communications, the navigator proceeds with an IMU fine alignment and the landmark navigational sightings. During these operations, the commander maneuvers the vehicle attitude for landmark acquisition. The navigator next compares his orbit information with ground computed parameters to ensure the accuracy of the onboard calculations. Thermal cycling will be accomplished as required during the 45-day mission.

The engineer prepares to transfer to the LEM for earth orbit operations. The commander checks the LEM pressurization from the CM. Then, the engineer equalizes the CM and interlock pressure, removes and stows the CM pressure and thermal

hatches, and disconnects and stows the probe and drogue. After activating and checking out his portable life support system, the engineer transfers from the CM to the LEM by proceeding through the tunnel. Upon completion of an inspection of the LEM cabin, he closes the latch and initiates communications with the MSFN. After a brief checkout period, experiments which require crew activation or participation are set up.

During each day of the orbit phase, the crew's primary activities include scientific experiments, system status checks, and IMU alignments. Personal functions are performed in the time remaining.

Nominally, while Vehicle A is in orbit 700, Vehicle B will be launched. Vehicle B enters its 80-n.mi. earth-parking orbit and prepares to transfer to its 200-n.mi. orbit for rendezvous with Vehicle A.

5.2.1.5 Phase 4: Rendezvous Preparations

Rendezvous preparations (see Appendix C) begin on the forty-third mission day of Vehicle A flight. The phase terminates with the first terminal maneuver.

On the forty-fourth day of Vehicle A orbit operations, the crew B enters Vehicle B in preparation for launch and subsequent rendezvous and docking with Vehicle A. Commander B arms the launch escape system, and crew members check out their pressure suits. The navigator and engineer then monitor the vehicle systems, and the mission controller makes the decision to launch based on previously selected launch constraints. Final countdown is entered, and launch is initiated.

5.2.1.6 Phase 5: Vehicle B Earth Ascent

The earth ascent phase (see Appendix C) begins with vehicle liftoff at T -0 and terminates with the S-IVB cutoff after parking orbit insertion.

Six seconds prior to liftoff, T -6 seconds, the S-IB is ignited. (The six seconds serve as a thrust buildup period.)

After liftoff, the S-IB* rises vertically (pitch = 90 degrees) for ( ) seconds and begins a preprogrammed pitch maneuver, which smoothly reduces the pitch angle to ( ) degrees at S-IB burnout. The peak dynamic pressure, ( ) pounds per square foot, is experienced at T +44 days, ( ) seconds at an altitude of ( ) feet. The time and altitude of the highest axial acceleration are ( ) seconds and ( ) feet. At S-IB burnout, at T +44 days, 141.64 seconds, the vehicle has a velocity of ( ) fps and an altitude of ( ) feet. The S-IB cutoff occurs 4.3 seconds prior to S-IVB ignition.

During the ascent of Vehicle B, the navigator monitors the S-IB ignition, the caution and warning indicator, and the environmental control system panel. For the duration of the ascent phase, the engineer maintains radio communication with the control center and monitors the critical temperature flow and events displays. The commander reports the S-IB engine cutoff and the S-IB/S-IVB separation, and continues to monitor acceleration, attitude, attitude rates, abort lights, and pitch and roll rates.

S-IVB ignition occurs at T +44 days, 145.94 seconds and thrusts for 449.6 seconds before cutoff. The vehicle pitch angle reduces from ( ) degrees at S-IVB ignition to ( ) degrees at the 80 n.mi. earth orbit insertion point.

After verifying S-IVB ignition, the commander continues to monitor vehicle performance. The commander confirms the LES jettison, 20 seconds after S-IVB ignition. Approximately 10 minutes after launch, the commander observes the S-IVB engine cutoff and vehicle insertion into a 80-n.mi. earth parking orbit.

5.2.1.7 Phase 6: Vehicle B Earth-Parking Orbit

The 80-n.mi. earth-parking orbit phase (see Appendix C) begins with S-IVB cutoff and concludes with SPS ignition for transfer orbit insertion.

With the termination of S-IVB thrusting, at a mission time of 596 seconds, the spacecraft crew performs post-injection checks and confirms trajectory parameters with MSFN. The spacecraft is then oriented for transposition and docking. After the S-IVB-SCS is placed in the attitude hold mode, the CSM/S-IVB-LEM separation devices are activated.

After S-IVB-LEM/CSM separation, the SLA is deployed to aid in the stabilization of the S-IVB-LEM and to expose the LEM. The CSM then translates 35 feet forward,

*No detailed launch trajectory print out is available at this time.

performs a 180-degree pitch maneuver, and translates back to the S-IVB-LEM. The CSM performs docking maneuvers to align the CM-LEM interface and to engage the LEM drogue (X-axis docking tunnel). A maximum of 15 minutes is required from separation to the accomplishment of a soft dock, which occurs at a mission time of ( ).

After soft docking is accomplished and the LEM and interlock pressurized, the CM hatches are opened, the hard dock latches are fastened, and the umbilicals are connected. Not more than 25 minutes is required from soft dock to the securing of the CM latches after performing the hard dock.

Once the CM latches are secured and preparations (including orientation) for S-IVB staging are completed, the separation devices are activated, initiating S-IVB (including IU and SLA)/CSM-LEM separation. Not more than 5 minutes is required from hard docking to the completion of S-IVB/CSM-LEM separation. S-IVB staging occurs at a mission time of ( ) at ( ) longitude and ( ) latitude.

Transposition and docking must be accomplished prior to a mission time of 4.5 hours, since the specified lifetime for the Saturn IB IU is 4.5 hours after liftoff.

Following S-IVB staging, the crew prepares for SPS insertion into the transfer orbit. IMU coarse and fine alignments, and navigational sightings, are made prior to the decision to continue the nominal mission. After the GNS is programmed, the spacecraft is properly oriented and the countdown for TOI initiated. At a mission time of 3 hours 10 minutes, the GNS automatically commands SPS ignition. SM RCS ullage ignition occurs 5 seconds prior to SPS ignition.

5.2.1.8 Phase 7: Vehicle B Transfer Orbit

The transfer orbit phase (see Appendix C) begins with the ignition of the SPS for transfer orbit insertion and terminates with SPS thrusting for earth orbit insertion. The nominal mission altitude is 200 n.mi.

During the second 80-n.mi. earth orbit, the SPS is used to transfer the spacecraft into a co-planar Hohmann transfer orbit with an apogee of 200 n.mi. The geocentric latitude and longitude of the point of thrust initiation are 21 and -83, respectively. The mission time is T +44 days, 3 hours, 10 minutes.

In preparation for the transfer orbit insertion, the navigator performs both coarse and fine IMU alignments, and then calculates the required transfer orbit parameters. The commander maneuvers the spacecraft attitude for the IMU alignment. This onboard trajectory data is compared with MSFN values, and the insertion point is established.

All three crewmen adjust their seat restraints. The commander makes a final check of the SPS and SM RCS for the transfer orbit burn and initiates the SPS ignition sequence. The three crewmen monitor the SPS ignition, burn, and cutoff.

The navigator confirms the transfer orbit by computing the trajectory parameters and comparing these onboard calculations with the MSFN values. The commander sets the controls for transfer orbit operations and checks the SPS parameters and propellant status.

During transfer orbit operations, the navigator performs a second IMU alignment in preparation for the second SPS burn into the 200-n.mi. orbit. During this period, the engineer checks out the electrical power and service propulsion systems. The commander maneuvers the spacecraft for the IMU alignment and maintains communication with the MSFN. The three crewmen then monitor the SPS ignition, burn, and cutoff.

5.2.1.9 Phase 8: Rendezvous of Vehicles A and B

The rendezvous phase of the mission (see Appendix C) begins at a mission time of T +44 days, 3 hours, and 55 minutes, with SPS cutoff for earth orbit insertion. Once the CSM A is hard-docked to the Spacecraft B/LEM A configuration the rendezvous phase ends.

After completing the SPS earth orbit insertion burn, the CSM subsystems are checked, and the orbit is confirmed with MSFN. A nominal launch of Vehicle B is assumed, and therefore, no maneuvers by Vehicle A are required.*

The final rendezvous and docking phase requires (1) the transfer of two crew members from CSM A to LEM A, (2) the separation of CSM A and LEM A, (3) the docking of LEM A and LEM B, and (4) the docking of CSM A to the resulting LEM A-LEM/CSM B configuration. Figure 5-1 shows the over-all sequence of docking maneuvers.

*In the trajectory analysis section some possible Vehicle A maneuvers are described to allow for off-nominal launches.

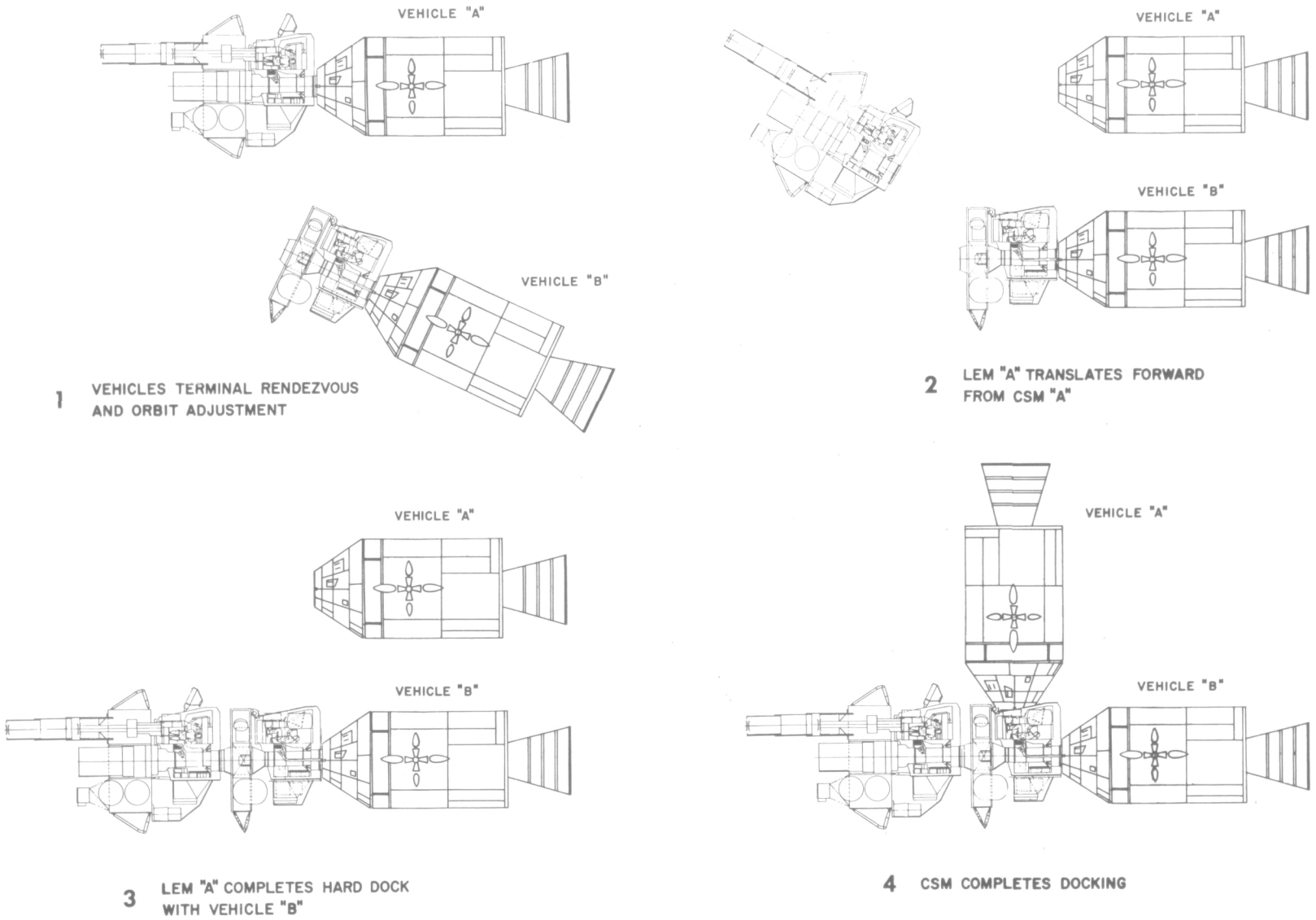


Figure 5-1. Terminal Rendezvous and Docking Sequence

After the navigators of Vehicles A and B have checked and verified the results of the final rendezvous thrust maneuver, the Vehicle A commander and engineer transfer to the LEM A in preparation for separation. The A crew members, including the navigator at the control panel in the CM, fasten their couch restraints. The A commander initiates LEM/CSM separation by igniting shaped charges to sever the LEM/CSM interface. Using the LEM A RCS, the commander translates LEM A from CSM A and performs the necessary attitude and translational maneuvers for a soft docking with the LEM B ascent stage docking ring. During the maneuver, the commander of Vehicle B maintains voice communications with LEM A for event and status verification. After soft docking has been achieved, B navigator secures the docking latches and electrical connections between the two LEM's and equalizes the cabin pressures.

LEM A now begins soft dock maneuvers with Spacecraft B. After soft docking is accomplished, the hard dock latches are fastened, umbilical connections made, and other hard-dock operations are completed.

The A navigator then performs a similar rendezvous maneuver to dock CSM A with the docking ring on the descent stage of LEM A. In both docking maneuvers, the A modules are the active vehicles, the CSM/LEM B configuration is passive. Both crews now prepare for personnel and equipment transfer.

Soft dock maneuvers between CSM A and Spacecraft B - LEM A are initiated. After engagement, hard dock proceedings are begun. With hard-dock completed, the two spacecraft enter the docked configuration coast phase.

5.2.1.10 Phase 9: Docked Configuration Coast (See Appendix C)

With the completion of the CSM A hard dock to Spacecraft B LEM A, the two spacecraft coast together in earth orbit until CSM A separates for deorbit. With the A and B spacecraft mated at the interface between the two LEM's, the crews initiate cargo transfer through the docking tunnels. This transfer includes the removal of active experiments from the CSM A to the LEM/CSM B and the replacement of extended LEM A equipment from Vehicle B supplies. Upon completion of this cargo transfer, the A commander and the B engineer join the A navigator in CSM A. The A engineer remains in the CSM-LEM A configuration with the replacement commander and navigator, thus one astronaut spends 90 days in earth orbit. Twenty-four hours after docking, both crews prepare for the separation and re-entry of the CSM A.

After all crew members have fastened their couch restraints, the A commander initiates CSM A separation by igniting shaped charges to sever the LEM A/CSM A interface. He then translates CSM A from LEM A using the SM reaction control system. The commander maneuvers CSM A to the proper attitude while the navigator performs a navigation sighting. After completing the IMU fine alignment and a landmark navigational sighting, the navigator computes the onboard trajectory. MSFN supplies orbital parameters for comparison to the navigator's trajectory calculations. Crew A then checks the CSM systems in preparation for a third SPS burn into the de-orbit coast phase.

CSM A separation occurs at mission T +45 days, 5 hours, 22 minutes; its position is ( ) longitude and ( ) latitude. Spacecraft B now enters its earth orbit coast phase, while CSM A begins de-orbit preparations.

5.2.1.11 Phase 10: CSM A De-orbit

The de-orbit phase (see Appendix C) begins when the crew starts de-orbit preparations and terminates with the CM entry into the earth atmosphere at 400,000 feet.

Preparations for CSM A staging are begun at a mission time of ( ). The crew installs and secures the hatches, orients the spacecraft for the separation maneuver, initiates the CM-LEM separation devices, and ignites the SM RCS to translate CSM A away from the LEM A Spacecraft B configuration. The CSM is separated at a mission time of T +45 days, 5 hours, and 22 minutes at a geocentric location of 200-n. mi. altitude, ( ) longitude, and ( ) latitude. MSFN tracking is available during this maneuver. CSM A separation nominally occurs prior to the last orbit.

After CSM A separation, the crew begins de-orbit preparations. The de-orbit maneuver parameters are determined with MSFN assistance and verification.

Following verification of the navigator's de-orbit parameters by MSFN, the three crewmen don their pressure suits and then adjust their seat restraints.

The CSM GNS is programmed and the CSM oriented before the crew initiates the countdown for SPS retro-ignition. At the termination of the countdown, the CSM GNS automatically commands SM RCS ullage ignition and SPS ignition at the proper times.

An SPS retrofiring of 10.5 seconds inserts the CSM into a descending transfer orbit having a vacuum perigee of 0 n.mi. The SPS thrust initiation begins at a mission time of ( ) and at 45 degrees longitude and -28 degrees latitude.

The commander checks the insertion parameters, and the CSM systems status, and communicates his findings to the MSFN. Next, he prepares for SM jettison by orienting the CSM to the proper attitude and activating the CM-SM separation sequence. During this period, the engineer switches all systems from SM to CM and retracts the antenna and all optics.

After a final IMU alignment, the CSM is properly oriented, and the SM is staged. SM staging occurs 10 minutes prior to re-entry. At the time of SM staging, the spacecraft is 190 miles above the earth's surface, at 110 degrees longitude and -12 degrees latitude.

All three crewmen monitor the CM-SM separation. After SM separation, the navigator and engineer perform a post-separation status check of the CM reaction control system. The commander then maneuvers the CM to the MSFN entry attitude. Following a check by the commander of the entry monitor flight indicator, the crew members adjust seat restraints, check suit umbilicals, and close face plates.

Following SM jettison, a final pre-entry check is conducted, which includes an RCS and GNS entry check, an entry monitor systems check, a crew equipment systems check, and a CM orientation for re-entry. The de-orbit phase ends when the CM re-enters the earth atmosphere at a nominal altitude of 400,000 feet. Re-entry occurs at a geocentric latitude of 8 degrees and a longitude of -145 degrees.

5.2.1.12 Phase 11: CM A Re-entry

The re-entry phase begins when the CM enters the earth atmosphere, nominally at an altitude of 40,000 feet, and ends at the time of drogue chute deployment, at an altitude of 24,000 feet.

At a mission time of T +45 days, 5 hours, and 49 minutes, a nominal entry altitude of 400,000 feet, and at a geocentric longitude and latitude of -145 degrees and 8 degrees respectively, the CM re-enters earth atmosphere.* For the preselected landing site of -156 degrees and 28 degrees, the range to go at the time of entry is ( ) n. mi. The entry velocity, flight path angle, inclination, and azimuth are 25,831 fps, -1.52 degrees, ( ) degrees, and ( ) degrees, respectively.

*No detailed re-entry printout is available at this time.

The initial part of the re-entry phase is a full-lift flight from 400,000 feet to a point where the rate of descent is reduced to 700 fps. At this point, trajectory range control is initiated. During trajectory range control, the steering commands are based on a reference trajectory, selected so that the guided trajectory predicted range from the control initiation point to touchdown matches the range requirements to reach the target. The final part of the entry phase consists of an equilibrium glide to an altitude of 24,000 feet, the altitude at which drogue chute deployment occurs.

During re-entry the commander continues to monitor the CM entry attitude and the guide maneuver. The navigator starts the elapse event time clock upon 0.05g indication. The engineer maintains communications with the MSFN recovery forces throughout the re-entry phase. At 100,000 feet, the commander arms the earth landing system and monitors the descent to 25,000 feet.

The forward heat shield is jettisoned at an altitude of 25,000 feet, three seconds prior to drogue chute deployment.

5.2.1.13 Phase 12: CM A Parachute Descent

The parachute descent phase begins at an altitude of 24,000 feet, at the time of drogue chute deployment, and ends approximately 7 minutes later with CM earth landing.

The navigator monitors the forward-heat-shield jettison at 25,000 feet and the drogue chute deployment 3 seconds later. If parachute deployment does not occur automatically, he deploys the chute manually.

The parachute descent phase begins at a mission time of T +45 days, 5 hours, and 59 minutes, approximately over the earth-landing site location. The CM is at a geocentric longitude of ( ) and a latitude of ( ) at the time of drogue chute deployment. The CM relative velocity and flight path angle at this point are ( ) fps and ( ) degrees, respectively.

The drogue chutes are jettisoned and main chutes are deployed at an altitude of about 11,000 feet, 58 seconds after the beginning of the phase.

After the navigator activates the VHF System, the three crewmen shut off all power switches not required for landing. The remaining propellant in the CM RCS is then

dumped by the navigator. The commander continuously monitors the altitude and descent velocity during this phase, communicating the status of the descent to MSFN.

Earth landing occurs at 420 seconds after the deployment of the drogue chutes. At CM-landing the relative velocity and flight path angle are 25 fps and -90 degrees respectively. Earth landing occurs at a longitude of -156° and a latitude of 28° in the Pacific Recovery Area (PRA).

Recovery communications (VHF) are activated and a final systems check is performed during parachute descent, starting 4 minutes and 49 seconds prior to touchdown.

5.2.1.14 Phase 13: CM A Recovery and Retrieval

The recovery and retrieval phase begins with CM touchdown and continues until both the CM and crew are on aboard a recovery ship.

The recovery and retrieval phase begins at a mission time of T +45 days, 6 hours, and 06 minutes, the time of the CM touchdown. Immediately after touchdown, each crew member releases his seat restraints and assesses his physical condition and the structural condition of the CM. While the commander activates the post-landing recovery aids, the engineer checks out the EPS and ECS for post-landing operations. After obtaining survival equipment from the storage area, crew A egresses to await recovery.

5.2.1.15 Phase 14: Spacecraft B/LEM A Earth Orbit Coast (See Appendix C)

The earth orbit coast period for the spacecraft B/LEM A configuration begins with the separation of CSM A from the docked configuration of the two spacecraft. Termination of this phase occurs at the time de-orbit preparations are begun. The mission duration and altitude are nominally 45 days and 200 n. mi.

Experiments that require crew activation or participation are initiated or set up according to the approved plan for experiments. The crew work/rest cycle is started. Periodic status checks of the CSM and LEM subsystem and periodic alignments of the IMU are accomplished on a scheduled basis. Thermal cycling will be accomplished as required throughout the mission.

On the last mission day, during the final orbits, preparations for de-orbit are begun by transfer of crew and material from the LEM's to CSM B. Phase 14 ends with these de-orbit preparations.

5.2.1.16 Phase 15: CSM B De-orbit

The de-orbit phase (see Appendix C) begins when the crew initiates de-orbit preparations and terminates with the CM entry into the earth atmosphere at 400,000 feet.

At the end of the second 45-day orbit phase of the mission, the engineer deactivates all LEM systems. The crew then transfers to LEM A and B all CM and experimental equipment to be jettisoned. Upon completion of these activities, each crew member enters his couch, fastens his restraints, and begins preparation for the jettison of the two LEM's.

Preparations for LEM A/LEM B staging are begun at a mission time of T +89 days. The crew orients the spacecraft for LEM staging, initiates the CM-LEM separation devices, and ignities the SM-RCS to translate the CSM away from the LEM A/LEM B configuration. The LEM's are staged at a mission time of T +89 days, 1 hour, and 30 minutes, at a geocentric location of 200 n. mi. altitude, ( ) longitude, and ( ) latitude. MSFN tracking is available during LEM staging. LEM staging nominally occurs prior to the last orbit.

After CSM LEM separation, the crew begins de-orbit preparations. The de-orbit maneuver parameters are determined with MSFN assistance and verification. The CSM GNS is programmed, and the CSM is oriented before the crew initiates the countdown for SPS retroignition. At the termination of the countdown, the CSM GNS automatically commands SM-RCS ullage ignition and SPS ignition at the proper times.

An SPS retrofiring of 10.5 seconds inserts the CSM into a descending transfer orbit having a vacuum perigee of 0 n. mi. The SPS thrust initiation begins at a mission time of T +89 days, 1 hour, and 30 minutes, and at 45 degrees longitude and -28 degrees latitude.

With the termination of SPS retro-thrusting, the crew accomplishes the final CSM subsystems status checks, coordinates with MSFN on re-entry, and begins preparations for the CM-SM separation. After a final IMU alignment, the CSM is properly oriented and the SM is staged. SM staging occurs 10 minutes prior to re-entry. At the time of SM staging, the spacecraft is 190 n. mi. above the earth's surface, at 110 degrees longitude, and -12 degrees latitude.

Following SM jettison, a final pre-entry check is concluded, which includes an RCS and GNS entry check, an entry monitor systems check, a crew equipment systems

check, and a CM orientation for re-entry. The de-orbit phase ends when the CM re-enters the earth atmosphere at a nominal altitude of 400,000 feet. Re-entry occurs at a geocentric longitude and latitude of -145 degrees and 8 degrees, respectively.

5.2.1.17 Phases 16, 17, and 18: CM B Re-entry, Parachute Descent, and Recovery

Re-entry, parachute descent, and recovery of CM B are exactly the same as CM A (see paragraphs 5.2.1.12 through 5.2.1.14), except for specific mission times.

5.2.2 CONFIGURATION

5.2.2.1 Introduction

The Command and Service Modules for both flights of Mission 5 are NAA Block II Modification D configurations (see 45). The Service Modules have an altered fuel loading, based on Grumman Aircraft Engineering Corporation studies (see 32), and an SPS retrofuel load was computed for this mission profile (see Part II, Section 6). The Command and Service Module changes - to become the Modification D configuration - are as shown in Table 5-6.

Table 5-6
Modifications to Block II CSM Systems

CSM System	Added to Block II	Removed from Block II
Communications and Data	Wiring and Wiring Interface	High-Gain and Rendezvous Transponder
Environment Control	Blower and Duct Compressor Space Cabin Lines 100-psi oxygen line Compressor Modified Cabin Fans	
Electrical Power SM Reaction Control	Wiring Interface Two sets LEM Tanks - per Quadrant	
Stabilization and Control	Horizon Sensors Redundant Units	

5.2.2.2 Spacecraft A LEM/LAB (See 33)

The spacecraft configuration and experiment locations for Flight A are shown in Figure 5-2 (see 33). The LEM/LAB includes ascent and descent stages.

Structural modifications to the LEM ascent stage are limited to the incorporation of an air lock below the floor panel in the forward cabin to accommodate a coronagraph. Support instrumentation is included with experiments carried on resupply flight. Supporting electronics and spares for experiments are carried in the aft equipment bay of the ascent stage.

The astronomy experiments are all aligned parallel to the longitudinal axis of the spacecraft, and are controlled from an astronomer console on the right side of the forward cabin. Access to the telescope and coronagraph is provided from inside the laboratory for readily changing cameras, magazines, and spectrographs. A common television viewfinder and two boresight star-trackers provide inputs for orientation and control. Spacecraft pointing accuracy for these experiments is ±1/2 degree. The telescope and coronagraph are gimballed and incorporate five error sensors to meet individual requirements.

Earth orientation is required for two weeks to accomplish the radio astronomy and cosmic ray emission experiments. During this period, the liquid/gas and solids behavior experiment apparatus is brought into the laboratory from outside stowage. After the experiments are completed, the apparatus is returned to external stowage. During this period, spacecraft stabilization input is provided by laboratory-mounted star trackers.

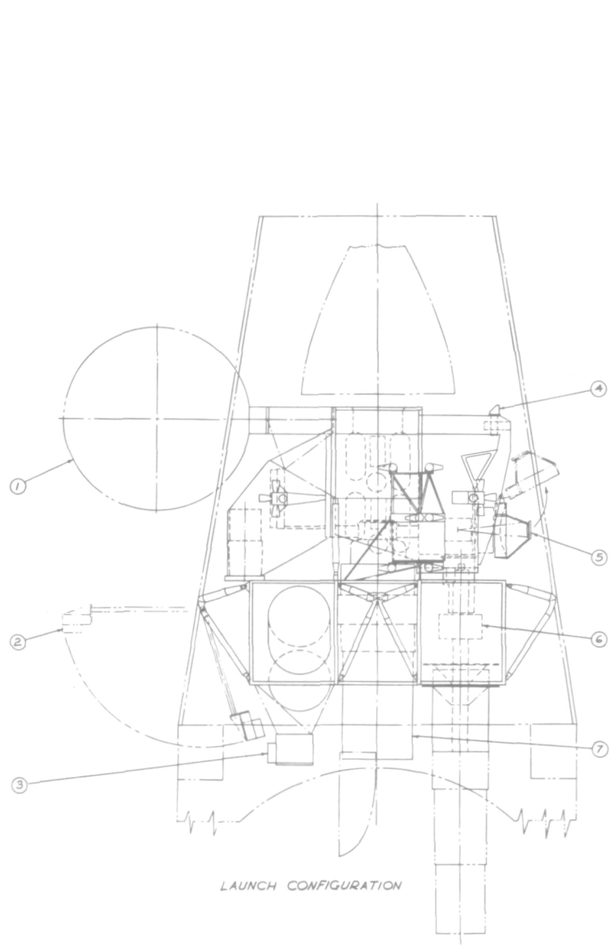
The radio astronomy antenna arrays, space structures experiment, and micrometeoroid collection apparatus are mounted externally on the descent stage. The radio astronomy antenna arrays are deployed manually during an EVA. The portion that remains attached to the spacecraft must be restowed upon completion of the experiment.

The telescope air lock is designed to mount in the existing ascent engine hatch. The descent stage structure must be reworked to accommodate the coronagraph. Supports for all other items can be added without modification.

Pointing and stability propellant requirements beyond the RCS propellant capacity of the CSM are accommodated by the laboratory. Additional propellant supply is

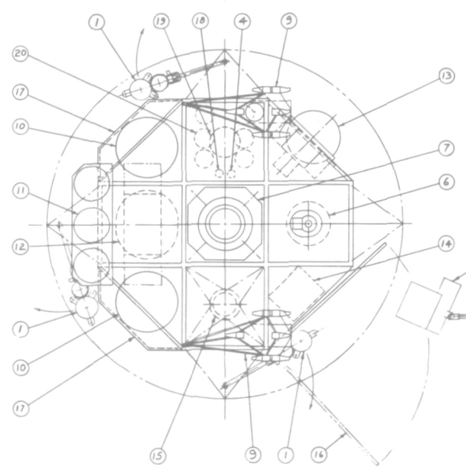
KEY

No.	Experiment	Name
1	0504	Micrometeoroid Collection
2	0606	Cosmic Ray Emulsion
3	0704	Study of Nearly Solar-Like Stars & X-Rays
4		Boresight Star Tracker
5	0605	Crystallization Studies
6	0703	Manned Coronagraph
7	0702	Intermed-Size Reflecting Telescope
8		RCS Fuel
9	0705A	Radio Astronomy 1-5 mc Range
10		H_2
11		FCA
12		O_2
13		
14		External Storage for Exp. 0601, 0602B, 0603A, 0603B, 0604, 0605 & Food.
15	0701	Emission Line Radiometry
16		FCA Radiator
17		Equipment Radiator
18		TV Viewfinder
19		GOX Tank
20		RCS Fuel (D Config. Only)
21		IR Earth Sensor

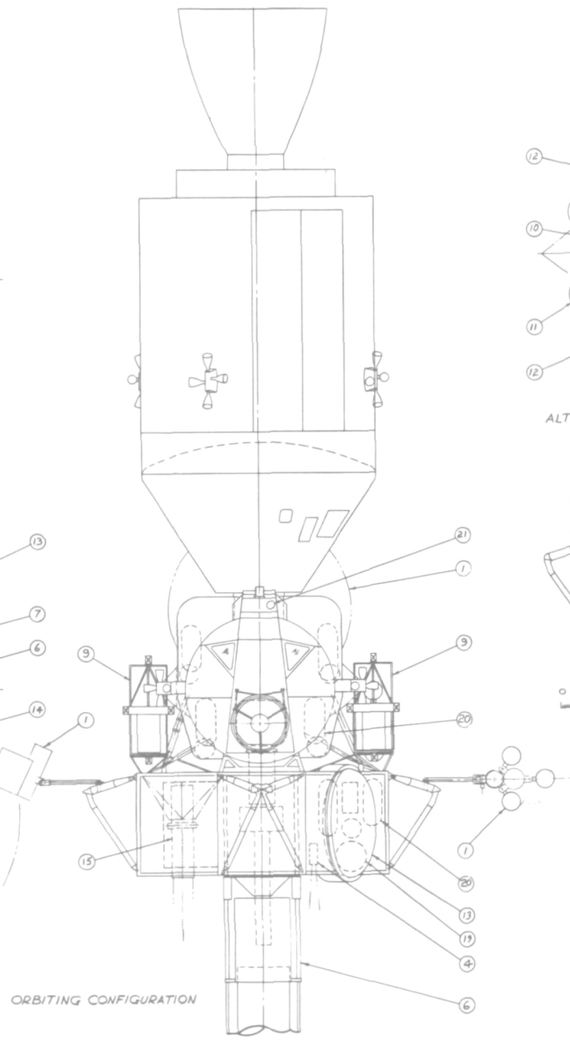


LAUNCH CONFIGURATION

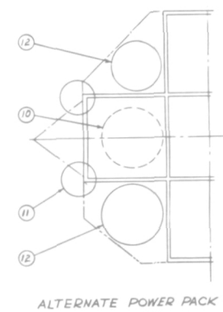
VIEW LOOKING DOWN AT ASCENT STAGE



VIEW LOOKING DOWN AT DESCENT STAGE



ORBITING CONFIGURATION



ALTERNATE POWER PACK CONFIGURATION

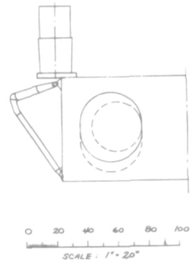


Figure 5-2. Spacecraft Configuration and Experiment Locations Flight A (Sheet 1 of 2)
(See 33 pages 1-13.38)

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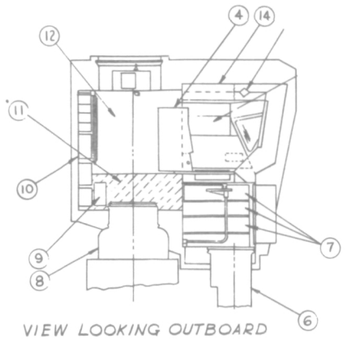
KEY

No.	Experiment	Name
1		Food
2		ECS
3		LiOH (10)
4		Biomed
5		
6	0703	Coronagraph
7		Data Recorder, Log & Data Book, Tape Recorder, Film & Tape Storage
8	0702	Telescope
9	0606	Cosmic Ray Emulsion
10		LGC
11		Equipt. Packages 0601, 0602B, 0603A, 0603B, 0604, & 0605. Held here during experiments. Upon completion will be removed to outside storage.
12	1401	Personnel Hygiene & Food Tech.
13		Floodlights
14	1401	Shower Stowed
15	0704	X-ray Photon Collector
16	0504	Micrometeroid Collection
17	0605	Solar Concentrator
18		Biomed Refrigerator
19		Biomed Incubator
20		
21		Astronomers Display Console
22	0602A	Liquid Drop Dynamics Study
23		Seat and Restraint (2)
24		Bicycle Ergometer
25		Attitude Controller Assy
26		Battery Charger, Pulse Height Analyzer, Tissue Equiv. Ionization Chamber
27		ECS Display Panel & Exp. 0403
28		N&G Display Panel
29	0504	Display Panel



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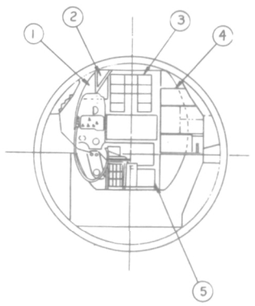
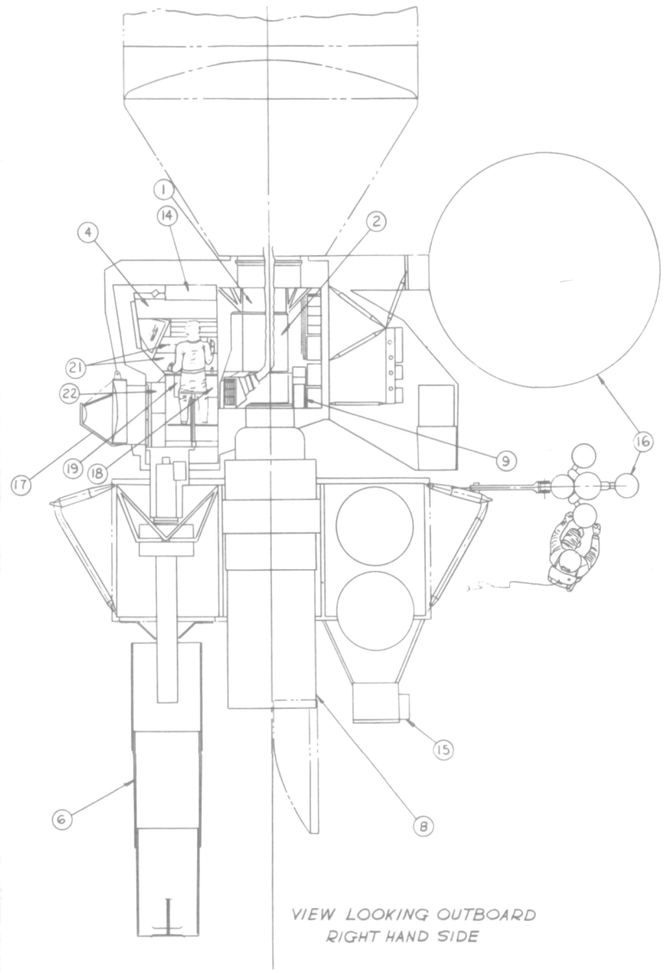
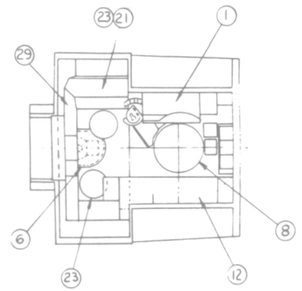
LEFT HAND SIDE



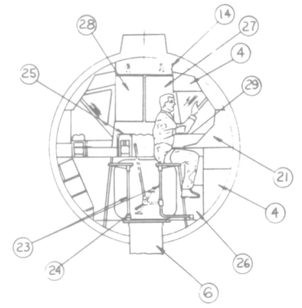
VIEW LOOKING AFT



VIEW LOOKING OUTBOARD
RIGHT HAND SIDE



PLAN VIEW



VIEW LOOKING FORWARD

0 20 40 60 80 100

Figure 5-2. Spacecraft Configuration and Experiment Locations Flight A (Sheet 2 of 2)
(See 33 pages 1-13.39)

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mounted on both the ascent and descent stages as required for the Modification D version.

In addition to the standard CSM/LEM interface, the initial flight requires additional wiring to accommodate the pointing requirements of the experiments. Interface wiring includes star tracker sensory inputs used to maintain attitude hold and command data from the laboratory flight controller for manual control of spacecraft slewing operations.

Adequate internal volume is available to accommodate experiments. A seven-day supply of lithium hydroxide and all but about six cubic feet of food is stored internally. Expended food and lithium hydroxide canisters are replaced from external storage by scheduled EVA's.

The flight utilizes an Apollo CSM fuel-cell electrical generation system, a LEM-ECS to provide independent life support operation during rendezvous, and select components to provide data transmission status and appropriate displays. VHF communications in the LEM provide voice and data links to the CM during rendezvous operations.

Conceptual sketches are available for four types of experiment equipment carried on this mission, Figure 5-3.

5.2.2.3 Spacecraft B

The spacecraft configuration and location of experiments for the second half of Mission 5 is shown in Figure 5-4.

The second flight, in addition to expendables and crew replacements, carries a laboratory for the study of living organisms, space structures technology, and extra vehicular operations. The laboratory is a LEM ascent stage, modified for the living organisms experiments. The experiments are located in the forward and center cabins. All animal containers are capable of pressurization, and are sustained by the existing LEM ECS, which is retained. Equipment for EVA's requiring internal storage is located on the aft bulkhead.

All food for the mission is carried internally and transferred to the food lockers of the previous flight after docking. Lithium hydroxide canisters are stored externally to replace internal storage on the A laboratory at seven-day intervals.

A 30-inch-deep, low-profile descent stage is utilized on this flight. This is necessary to provide staging clearance for the large-aperture space-erectable antenna experiment. A 44-inch diameter pressurized passageway is mounted in the center bay of the rack, the upper end of the passageway is designed to fit into the engine hatch of the ascent stage laboratory pressure hatch provided for it. The opposite end of the passageway duplicates the docking part of the CM, providing compatible docking procedures with the previous flight.

The laboratory is used as an airlock for EVA; and apparatus associated with the EVA experiment is mounted on the descent stage convenient to the forward hatch.

The large-aperture space-erectable antenna experiment is suspended below the descent stage during launch, and manually moved to deployment position during an EVA. The additional attitude-hold propellant requirement is satisfied by adding the LEM RCS system and tankage.

All experiment equipment installations involve additions to the existing LEM ascent stage and basic low-profile descent stage structure.

In addition to the standard CSM/LAB interface, this flight requires additional interface connections between this laboratory and the originally resupplied laboratory. The interface wiring is the same as that required for the laboratory/CSM interface.

This flight utilizes an Apollo CSM fuel-cell electrical generation system, a LEM ECS to provide atmosphere revitalization and waste control to the animal tanks and recirculation of cabin atmosphere, and selected components to provide data transmission, status and appropriate displays.

5.2.2.4 Configuration History

The configuration history for Mission 5 is shown in Figure 5-5, which divides the mission profile into five major operations areas. The launch vehicle and spacecraft configurations are depicted for each major area. Mission times for the trajectory analysis and mission phasing charts are provided to relate time changes.

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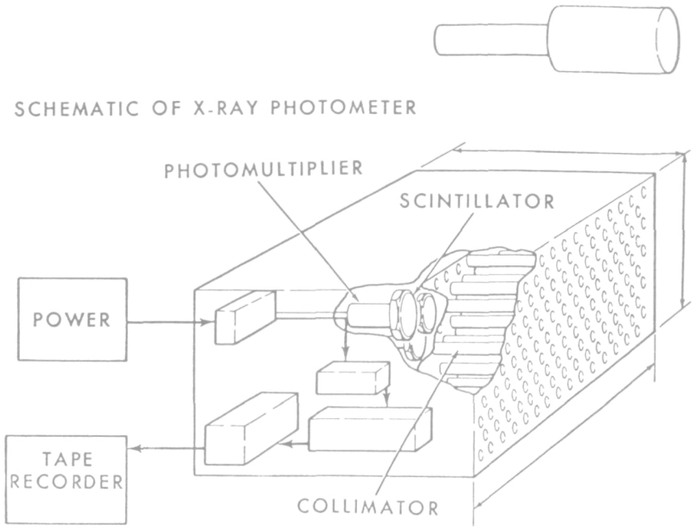
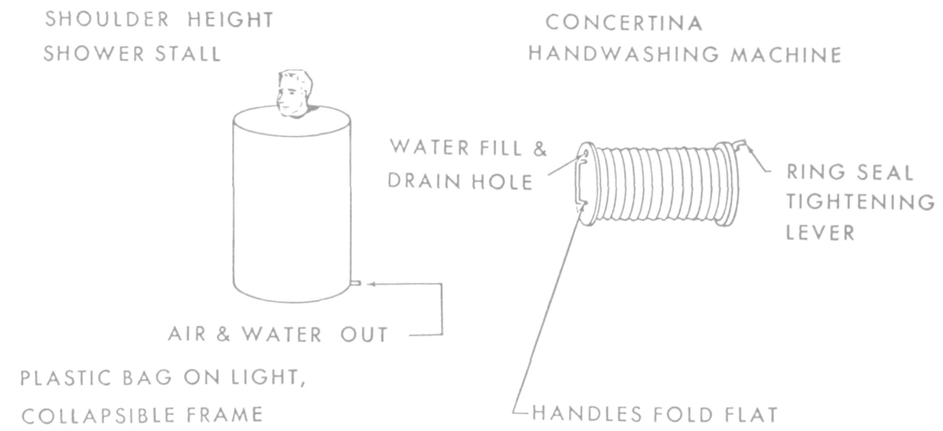
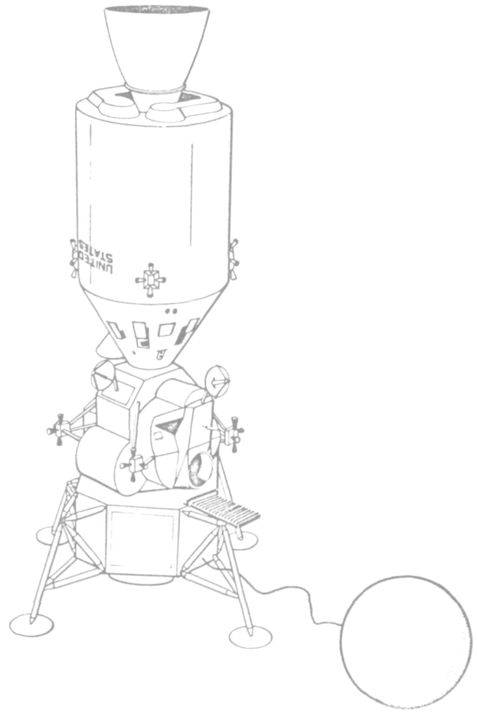


PERSONAL HYGIENE & FOOD TECH - 1401



DEPLOY RF REFLECTIVE STRUCTURES - 1302



MICROMETEOROID COLLECTION - 0504

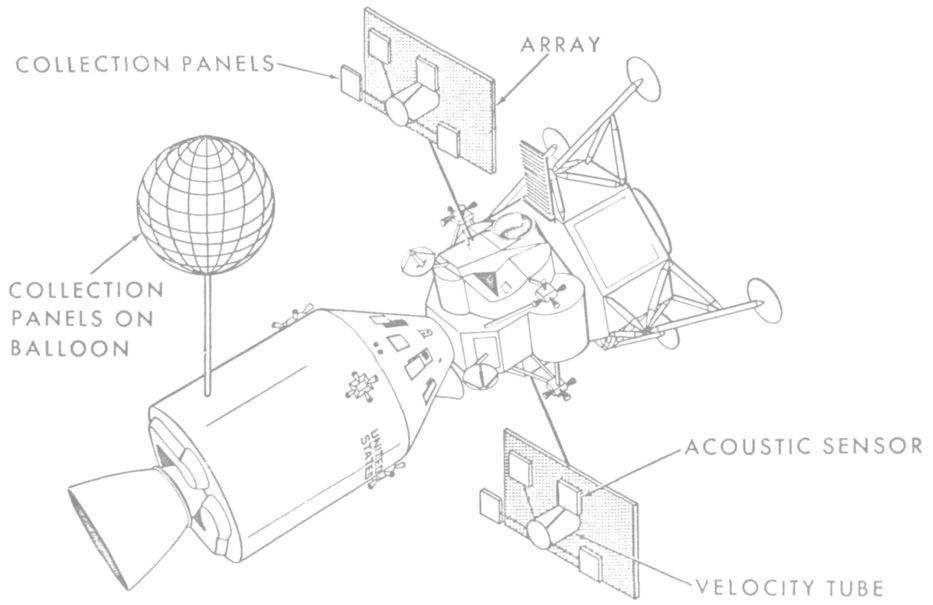
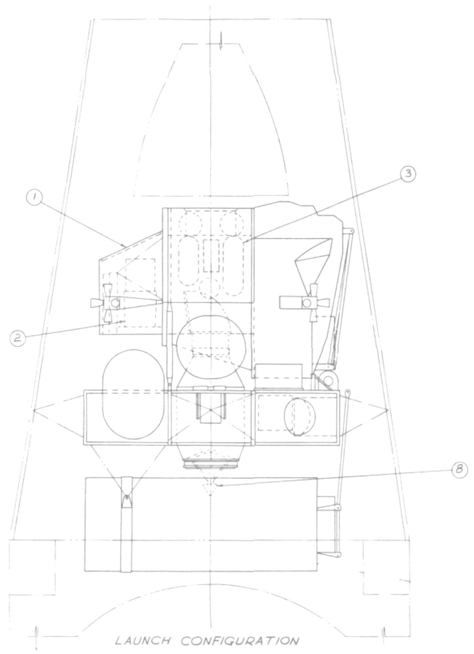
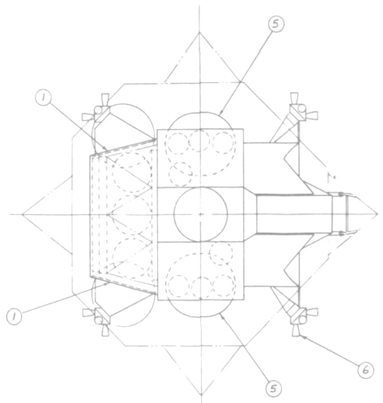


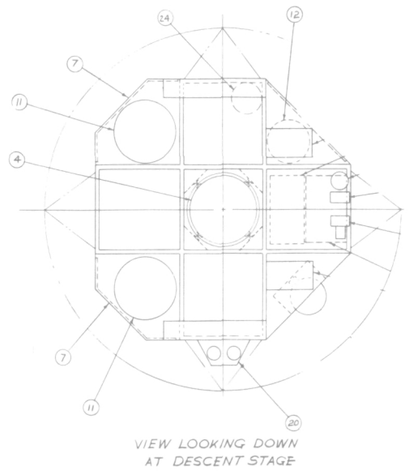
Figure 5-3. Experiment Concept Sketches (See 11, 13)

KEY

No.	Experiment	Name
1		FCA Radiator
2		FCA
3		RCS Fuel ("D" Config. only)
4		Pressurized Passageway
5		O_2
6		RCS Thrusters ("D" Config. only)
7		Equipt. Radiator
8		Docking Probe (Stored in CM)
11		H_2
12		Water
20	0406	Microorganisms Collection Plates
21		LioH (36 CM Canisters)
22		Water ("C" Config. only)
23		Flight 13 Lab.
24		GOX Tank







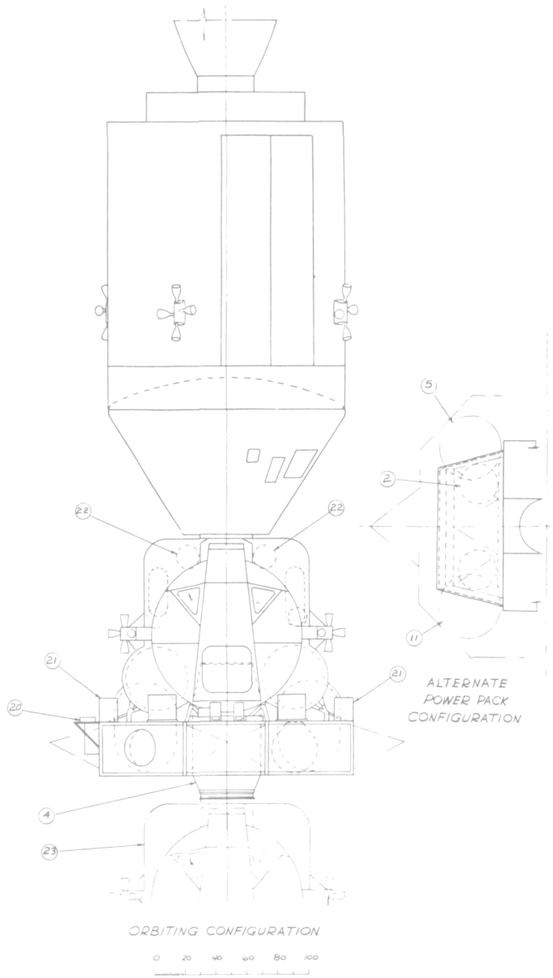
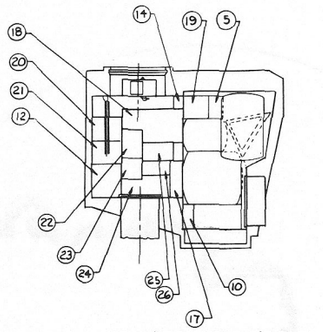


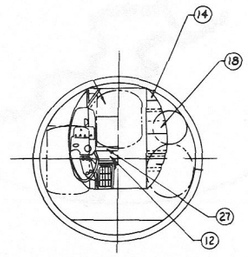
Figure 5-4. Spacecraft Configuration and Experiment Location (Sheet 1 of 2)
(See 33 pages 1-14.39)

KEY

No.	Experiment	Name
1		Film Storage
2	0401&0402	Incubator
3	0406	Incubator
4	0405	Food Storage
5	0404	Equipt. Storage
6	0404	Planaria & Newt
7	0405	Rats
8	0403	Chimpanzee Instr. Panel
9	0403	Chimpanzee
10	0403	Waste & Food Management
11	0400	Freezer
12	0402	-20°C Refrigerator
13	0404	
14	0401	Lyophilization
15	0405	Waste Storage
16	0405	Quick Freezer
17	0401	Chemical & Equipt. Storage
18	0402	Culture Flasks
19	0403	Supply Cabinet
20	0406	+5°C Refrigerator
21	0404	-5°C Refrigerator
22	0402	Particle Counter
23	0402	Centrifuge
24	0402	Supply Cabinet & 35mm Camera
25	0402	Scintillation Counter
26	0401,0402 0406	Microscope



VIEW LOOKING OUTBOARD
LEFT HAND SIDE



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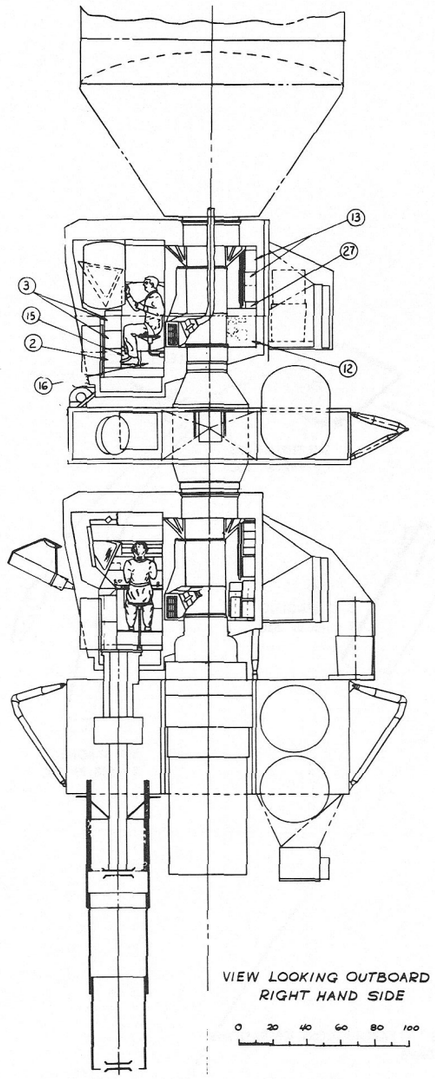
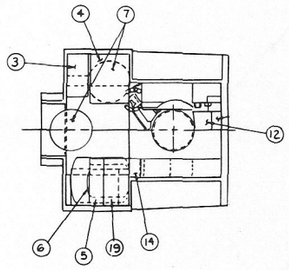
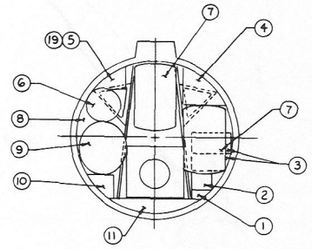


VIEW LOOKING OUTBOARD
RIGHT HAND SIDE

0 20 40 60 80 100



PLAN VIEW



VIEW LOOKING FORWARD

Figure 5-4. Spacecraft Configuration and Experiment Location
(Sheet 2 of 2)
(See 33 pages 1-14.40)

5-45-1

2

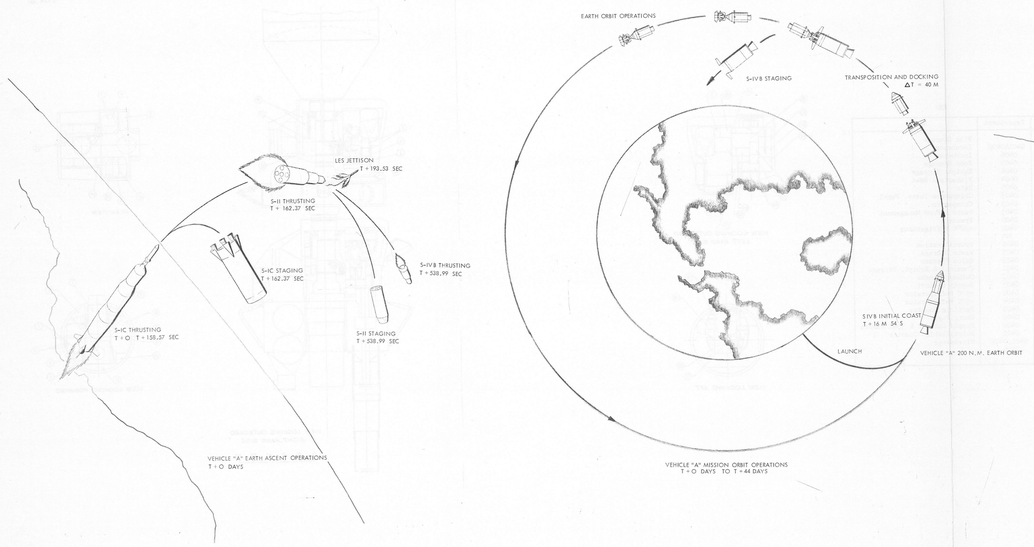
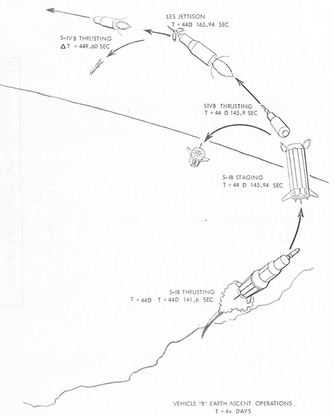


Figure 5-5. Mission 5 Configuration History

5-46-2

MISSION 5 S-V/S-1B RENDEZVOUS



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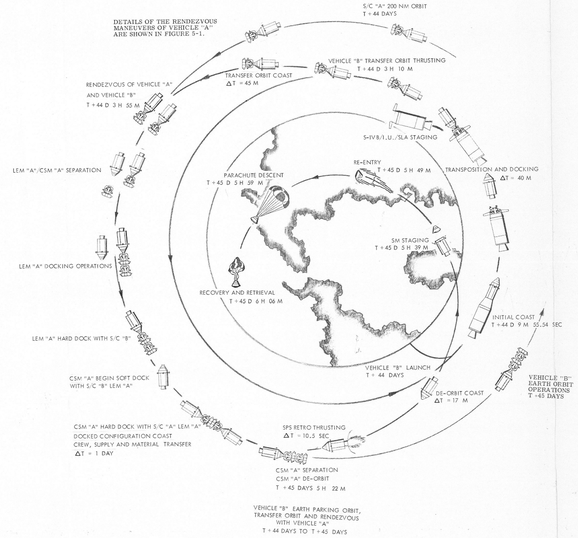

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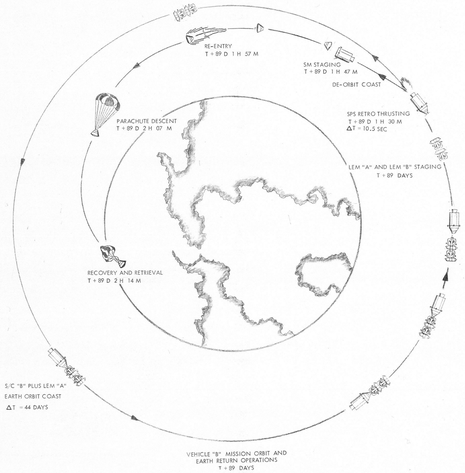

5-46-5

5.2.2.5 Expendables Weights - Vehicle A

5.2.2.5.1 *Life Support System*

The expendable weights for Mission 5, Vehicle A, are summarized in Figure 5-6. The following expendables weights are ascribed to the life support system (see 126 and 127):

a. Food - 228 pounds are stored on the LEM (see 33-1-13.36). This amount is sufficient to sustain the crew for the mission at a consumption rate of 1.4 pounds per man per day (see 126).

b. Oxygen - The CSM has two oxygen tanks with a capacity of 652.8 pounds. The LEM has two Apollo oxygen tanks (see 33-1-13.15) loaded to 2382 pounds.

c. Hydrogen - The CSM has two hydrogen tanks filled to 58.24 pounds. The LEM has two Apollo hydrogen tanks (see 33-1-13.15) loaded to 226 pounds.

d. Repressurization - For extravehicular activity, the CM atmosphere must be evacuated overboard. It was assumed that the CM and LEM have no airlocks and the LEM atmosphere is also dumped. Repressurization requires six pounds of oxygen for the LEM, and seven pounds for the CM (see NAA Training Manual). Seven repressurizations are required (see 33-1-13.14).

e. Leakage - The oxygen leakage rate for the CM is 5.66 pounds per day for the LEM 4.8 pounds a day (LEM criticality design reference). This results in total oxygen leakage of 216 pounds for the LEM and 255 pounds for the CM, based on a cabin pressure of 5 psia.

f. Respiration - The crew oxygen requirement is 1.67 pounds per man per day, or a total of 2255 pounds to be supplied by the LEM.

g. Fuel Cells - The CSM and the LEM each have three fuel cell assemblies (see 80 and 33-1-13.15). The hydrogen and oxygen supply for the CSM cells is the original supply less the amounts required for respiration repressurization and leakage replacement. The fuel cell production of water varies, with power requirements, from 28.8 to 82.8 pounds per day for the three assemblies combined. Hydrogen and oxygen available to the fuel cells combine to form water. Any excess water produced by the cells is vented overboard.

h. Water - To reconstitute the powdered food, 0.035 pounds per man per hour of water is required, for a total of 113 pounds. For drinking,

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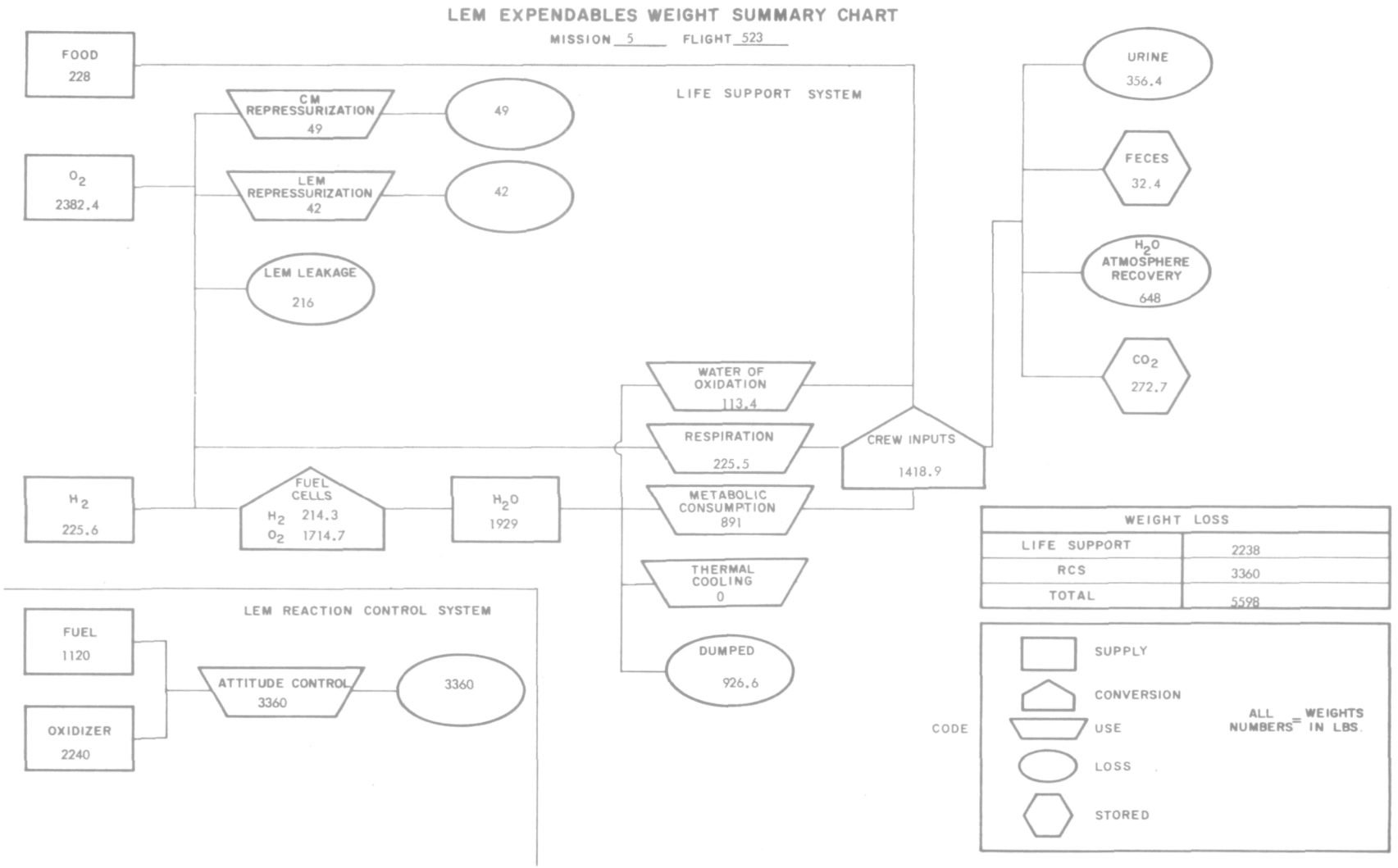

Figure 5-6. Expendables Weight Summary Chart Flight 523 (Sheet 1 of 2)

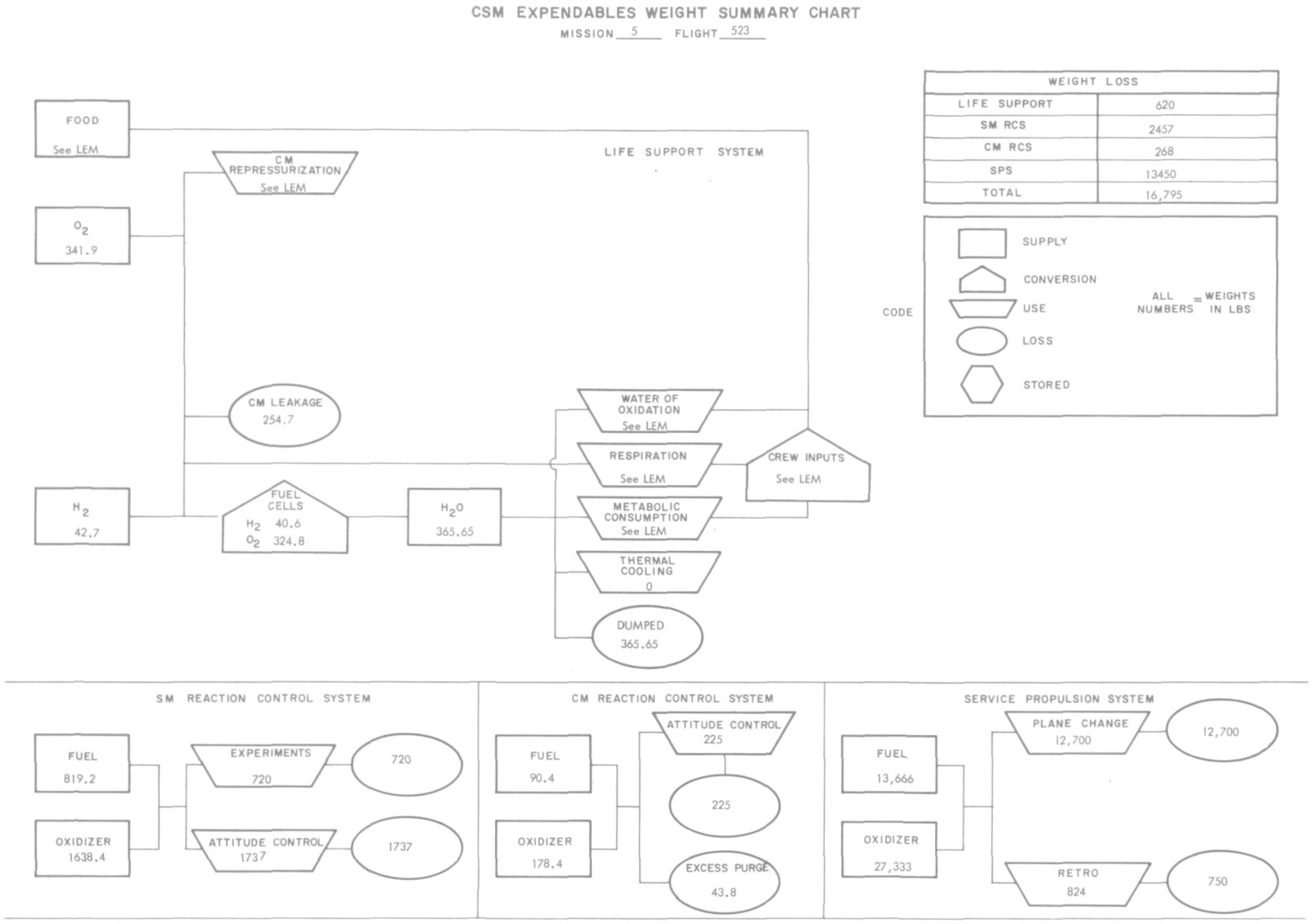

Figure 5-6. Expendables Weight Summary Chart Flight 523 (Sheet 2 of 2)

0.275 pounds per man per hour is required, for a total of 891 pounds. No LEM cooling water is required (see 33-1-13.15); the same was assumed for the CM.

i. Human Waste - Urine is produced at a rate of 0.11 pounds per man per hour, for a total of 356 pounds, which is vented overboard. Feces is produced at a rate of 0.24 pounds per man per day, totaling 32.4 pounds, which is stored aboard.

Water vapor, from expiration and perspiration, is given off at the rate of 0.2 pounds per man per hour, totaling 648 pounds, which is recovered and dumped. Carbon dioxide, from respiration, is produced at a rate of 2.02 pounds per man per day (see SID 63-1370-2). The total amount of 32.4 pounds is stored on board, absorbed by lithium hydroxide filters.

5.2.2.5.2 SM RCS

The two LEM SM RCS fuel tanks per quad are loaded to 819 pounds. The two LEM SM RCS oxidizer tanks per quad are loaded to 1638 pounds.

The RCS requirements for attitude control during the experiments is 720 pounds of fuel-oxidizer mixture in a ratio of one-to-two. The remaining reactants are available for normal attitude control functions, and for:

a. CSM booster separation.
b. Ullage maneuver prior to SPS ignition.
c. Separation of SM and CM.

5.2.2.5.3 CM RCS

The two CM RCS fuel tanks are filled to 90.4 pounds. The two CM RCS oxidizer tanks are loaded to 178.4 pounds. A 225-pound load of reactants provides:

a. Maneuver capabilities during entry after SM separation.
b. Maneuver capabilities in the event of a high-altitude abort after LES jettison.
c. Three-axis control to stabilize CM motion prior to deployment of the ELS parachutes.

Fuel and oxidizer left in the tanks after parachute deployment and final attitude correction are dumped before touchdown.

5.2.2.5.4 SPS

The single SPS fuel tank has a capacity of 15,735 pounds; the load for this flight is 13,666 pounds. The one SPS oxidizer tank has a 31,620-pound capacity; the load for this flight is 27,333 pounds.

A reactant weight of 12,700 pounds is required for plane changes in preparation for rendezvous; 750 pounds is required for the retro maneuver (calculated). (For minimum SPS loading, see Part II, Section 6.)

5.2.2.5.5 LEM RCS

The twelve LEM RCS fuel tanks (see 33-1-13.15) have a 1226-pound capacity; the load for this flight is 1120 pounds. The twelve Block II LEM RCS oxidizer tanks (see 33-1-13.15) have a 2457-pound capacity; the load for this flight is 2240 pounds. LEM attitude control required 3360 pounds of reactants (see 33-1-13.36).

5.2.2.6 Expendables Weights - Vehicle B

5.2.2.6.1 Life Support System

The expendable weights for Mission 5 are summarized in Figure 5-7. The following expendable weights are ascribed to the life support system (see 126 and 127):

a. Food - 228 pounds are stored on the LEM (se 33-1-13.24), sufficient to sustain the crew for the mission at a consumption rate of 1.4 pounds per man per day.

b. Oxygen - The CSM has two oxygen tanks with a total of 652.8 pounds (see 80-167). The LEM has two Apollo X tanks (see 33-1-14.13) loaded to 2,291 pounds.

c. Hydrogen - CSM has two hydrogen tanks loaded to 58.24 pounds (see 80-167). The LEM has two Apollo X-tanks (see 33-1-14.13) loaded to 208 pounds.

d. Repressurization - For extravehicular activity, the CM atmosphere must be evacuated overboard. It was assumed that the CM and LEM have no airlocks and that the LEM atmosphere is also dumped. Repressurization requires 6 pounds of oxygen for the LEM; 7 pounds for the CM (see NAA Training Manual). Seven repressurizations are required (see 33-1-14.13).

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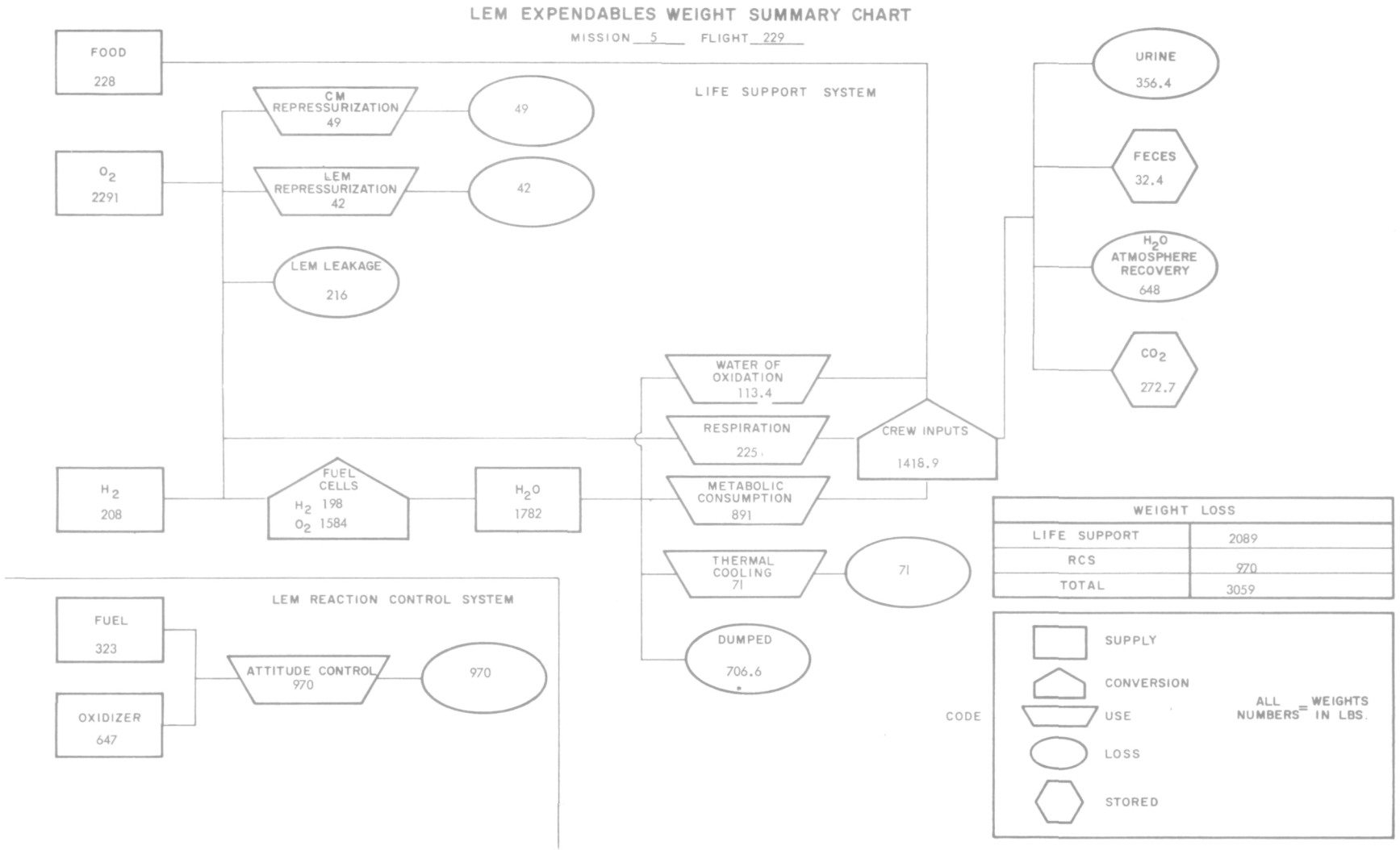

Figure 5-7. Expendables Weight Summary Chart Flight 229 (Sheet 1 of 2)

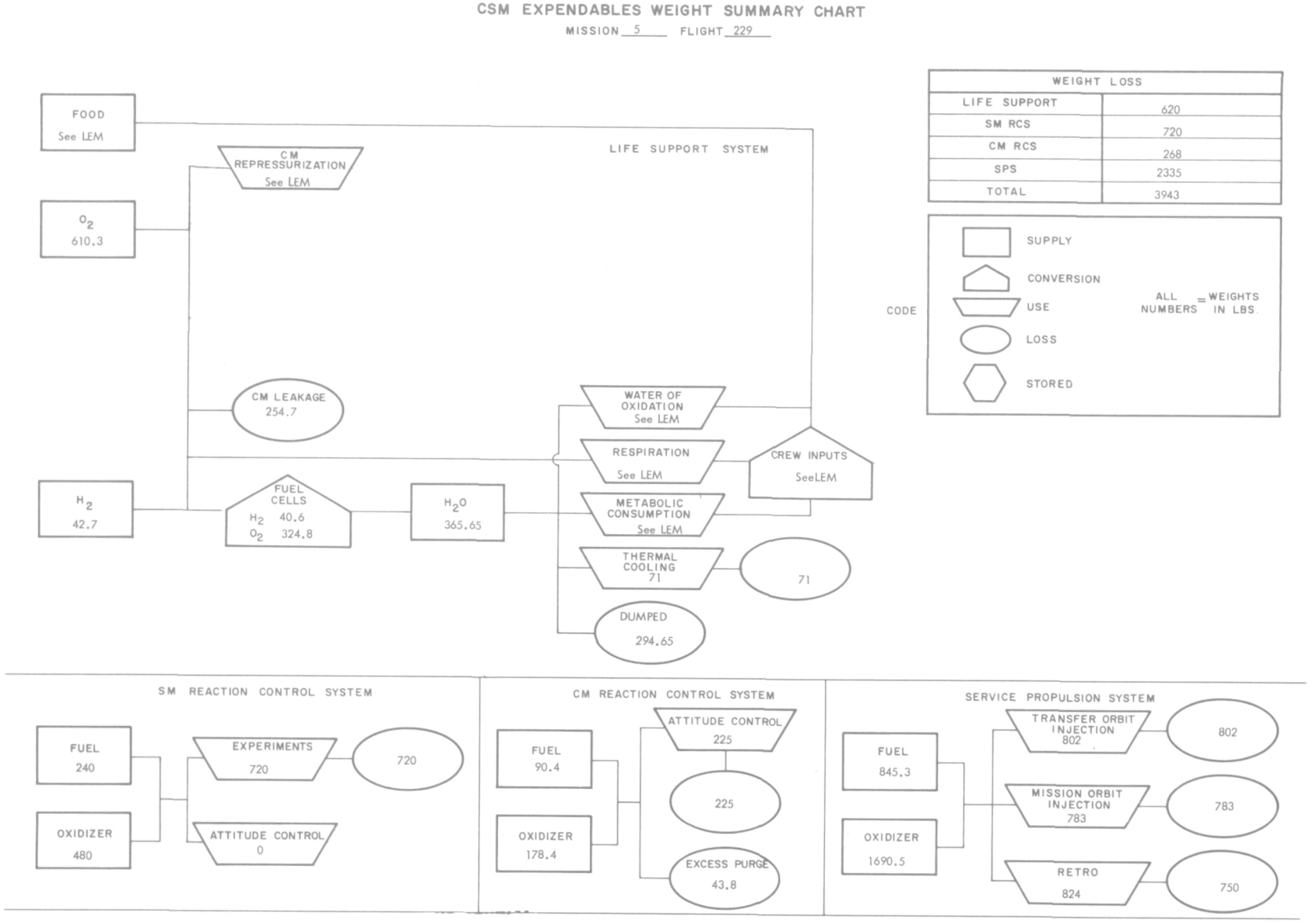

Figure 5-7. Expendables Weight Summary Chart Flight 229 (Sheet 2 of 2)

e. Leakage - The oxygen leakage rate for the CM is 6.55 pounds per day; for the LEM, 418 pounds per day. This results in a total oxygen leakage of 216 pounds for the LEM and 225 pounds for the CM, based on a cabin pressure of 5 psia.

f. Respiration - The crew oxygen requirement is 1.67 pounds per man per day, requiring that 225 pounds be supplied by the LEM.

g. Fuel Cells - The CSM and the LEM each have three cell assemblies (see 80 and 33-1-14.13). The hydrogen and oxygen supply for the CSM cells is the original supply less the amounts required for respiration, repressurization, and leakage replacement.

h. Water - The fuel cell production of water varies, with power requirements, from 28.8 to 82.8 pounds per day for the three assemblies combined. Hydrogen and oxygen delivered to the fuel cells combine to form water.

i. Water of Oxidization - Water is needed to reconstitute the powdered food. The requirement is 0.035 pounds per man per hour for a total of 113.4 pounds.

j. Metabolic Consumption - Drinking water 0.275 pounds per man per hour for a total of 891 pounds.

k. Thermal Cooling - LEM cooling requires 71 pounds of water (see 33-1-14.13). An equal amount was assumed for the CM.

l. Water Dumped - Excess water produced by the fuel cells is vented overboard.

m. Urine - Urine, as human waste, is produced at a rate of 0.11 pounds per man per hour, totaling 356 pounds, vented overboard.

n. Feces - Human feces waste is produced at a rate of 0.24 pounds per man per day. The total amount is 32.4 pounds, stored aboard.

o. Water Atmosphere Recovery - Water vapor from expiration and perspiration is given off at the rate of 0.2 pounds per man per hour. The water vapor, a total of 648 pounds, is recovered and dumped.

p. Carbon Dioxide - Human respiration waste is produced at a rate of 2.02 pounds per man per day (see SID 63-1370-2) for a total of 273 pounds, which is stored on board, absorbed by lithium hydroxide filters.

5.2.2.6.2 SM RCS

There are two LEM RCS fuel tanks per quad with a capacity of 819 pounds. There are two LEM RCS oxidizer tanks per quad with a capacity of 1638 pounds.

The RCS requirements for attitude control during the experiments of 720 pounds of fuel-oxidizer mixture in a ratio of one to two. The remaining reactants are available for normal attitude control functions, and for:

a. CSM booster separation.
b. Ullage maneuver prior to SPS ignition.
c. Separation of SM and CM.

5.2.2.6.3 CM RCS

There are two CM RCS fuel tanks loaded to 90.4 pounds. There are two Block II CM RCS oxidizer tanks loaded to 178.4 pounds. A 225-pound quantity of reactants provides:

a. Maneuver capabilities during entry after SM separation.
b. Maneuver capabilities in the event of a high-altitude abort after LES jettison.
c. Three-axis control to stabilize CM motion prior to deployment of ELS parachutes.

Fuel and oxidizer left in the tanks after parachute deployment and final attitude correction is dumped before touchdown.

5.2.2.6.4 SPS

The single SPS fuel tank has a total capacity of 15,735 pounds; the fuel load for this flight is 845 pounds. The single SPS oxidizer tank has a 31,260-pound capacity; the oxidizer for this flight is 1690 pounds.

There is 802 pounds of reactants required for transfer orbit injection (see 111); 738 pounds of reactant for mission orbit injection (see 111); and 750 pounds for the retro maneuver (calculated).

5.2.2.6.5 LEM RCS

The four LEM RCS fuel tanks (see 33-1-14.13) have a 406-pound capacity, the load for this flight is 323 pounds. The four LEM RCS oxidizer tanks have a 1819-pound capacity; the load for this flight is 647 pounds.

LEM attitude control requires 970 pounds of reactants (see 33-1-14.24).

5.2.2.7 Weight Profile

5.2.2.7.1 General

The vehicle weight changes for Mission 5, beginning with S-IVB burnout of each flight through CM re-entry are shown in Figure 5-8. Data for the chart were taken from Reference 111 and from the Weight Summary Chart (Tables 5-1 and 5-2), and the Expendables Weight Summary Charts (Figures 5-6 and 5-7).

5.2.2.7.2 Vehicle A

For this flight, the vehicle weight at S-IVB burnout is 273,495 pounds. S-IVB staging, and jettisoning the LEM adapter reduces the weight to 232,956 pounds.

The payload weight for this flight is 87,510 pounds. The differential between maximum vehicle weight and payload weight is shown as "margin," which for this flight is 22,490 pounds.

During the forty-four day, earth orbit coast, the consumption of expendables reduces the spacecraft weight to 78,835 pounds. SPS plane change thrusting brings the weight to 66,135 pounds. At this time Vehicle A is docked to Vehicle B.

5.2.2.7.3 Vehicle B

The weight of Vehicle B at S-IVB burnout is 71,255 pounds. S-IVB staging and jettisoning the LEM adapter reduces the weight of 36,270 pounds. Fuel expenditure for transfer orbit injection and mission orbit injection brings the weight down to 34,685 pounds.

The payload weight required for this mission is 32,720 pounds. The differential between the required and available payload weights is shown as a margin, which for this mission is 1965 pounds.

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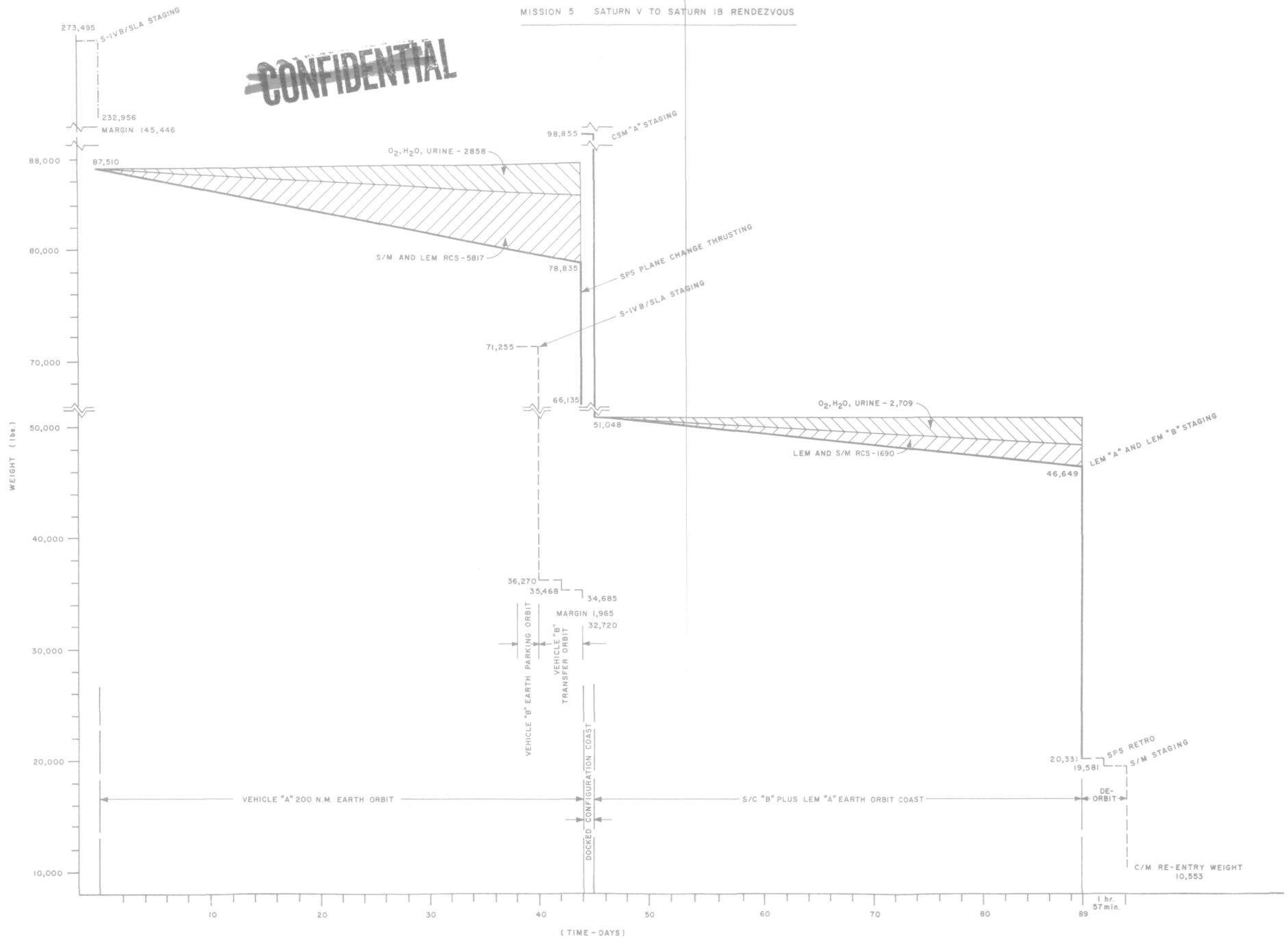

Figure 5-8. Weights Profile

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5.2.2.7.4 Vehicles A and B Docked

The combined docked weight of the vehicles is 98,855 pounds. The CSM of Vehicle A is staged and returns to earth (re-entry weight is 10,553 pounds).

This reduces the payload weight in orbit to 51,048 pounds. During the next 45 days in orbit, the consumption of expendables reduces the weight of 46,649 pounds.

Staging the A and B LEM's brings the weight to 20,331. Retro-fire fuel consumption and SM staging reduces the re-entry weight of CM B to 10,553 pounds.

5.2.3 CREW ANALYSIS

5.2.3.1 Time Utilization

Crew tasks performed during Mission 5 fall into two basic categories: flight operations and experiments. The crew flight operations required from liftoff to the 200-n. mi. orbit, and from initiation of de-orbit to touchdown, are detailed in Appendix D. Flight operations activities required of the crew during the 200-n. mi. orbit consist primarily of system status checks, IMU alignments, and communications with MSFN. Table 5-7 summarizes the system status checks adapted from the Apollo design reference mission. A detailed task analysis of the crew activities required for experiments is also presented in Appendix D.

Analysis of the flight crew operations, and the sleep, eating, hygiene, and rest requirements for a typical orbital day indicate that approximately 24 man-hours are available in a 24-hour day to perform experiments. This allows a total of approximately 1080 man-hours available for experiments over the 45 days of Flight 523. Since only the first three days of Flight 229 were scheduled, no attempt was made to project that flight for 45 days. Utilizing the number of experiment sequences and the number of test subjects (see 31, 32, 33) recommended for the assigned experiments with the experiment times as outlined in Appendix D, it was determined that approximately 1699 man-hours would be required to conduct the assigned experiments. Therefore, an experiment priority system was established.

5.2.3.2 Experiment Priorities

5.2.3.2.1 Introduction

The priority classifications scheme and priority numbers are as follows:

a. First priority experiments are those that satisfy primary mission objectives.
b. Second priority experiments are those that contribute to the assessment of physical health or well being of the astronauts.
c. Third priority experiments are those that satisfy secondary mission objectives.
d. Fourth priority experiments are the biomedical and behavioral experiments, which contribute general scientific knowledge.

e. The lowest priority experiments are those that have been performed on previous flights, and are of such a nature that their omission is not detrimental to mission objectives.

It should be noted that the priority classification scheme and assignments were arbitrarily made due to the excessive experiment time load. They do not constitute a recommended or suggested solution to the experiment priority and integration problem.

Table 5-7

Breakdown of
Earth Orbital System Status Checks*

Task	Frequency of Performance	Duration (Minutes)
CM		
Check ECS	Hourly	1
Check EPS	Hourly	3
Check SM RCS	Every 4 hours	4
Check CM RCS	Every 12 hours	4
Replace CO_2 and odor absorber	Every 12 hours	8
Charge entry and post-landing batteries	Every 6 hours	3
Purge fuel cells	Every 8 hours	3
Check SPS	Every 4 hours	4
LEM		
Check ECS	Hourly	1
Check PGS	Hourly	1
Check PDS	Hourly	1
Replace CO_2 and odor absorber	Every 12 hours	8
Check LEM-RCS	Every 12 hours	4
4.75 man-hours required for system status checks in a 24-hour day.		

*AMPTF adapted (24, 25, 26)

CM tasks taken from translunar and transearth coast phase.

LEM tasks taken from lunar surface phase.

5.2.3.2.2 Flight A

Experiment priorities for the first flight of Mission 5 are as follows:

Priority	Experiment Number and Group	
1	0700 Series	Astronomy - 0701, 0702, 0703, 0704, 0705A
2	0100 Series	Biomedical - 0109, 0110, 0111, 0112, 0113, 0114, 0118, 0115, 0116, 0119, 0120,
2	0501	Radiation Monitoring
3	0600 Series	Zero-g - 0601, 0602A, 0602B, 0603A, 0603B, 0604, 0605, 0606
3	1302	RF Reflective
3	0504	Meteorical
3	1504	Transfer Operations
3	1401	Hygiene and Food
4	0100 Series	Biomedical - 0103, 0104, 0105, 0106, 0107, 0108, 0117
4	0200 Series	Behavioral - 0201, 0202, 0203

5.2.3.2.3 Flight B

Experiment priorities for the second half of Mission 5 are as follows:

Priority	Experiment Number and Group	
1	0400 Series	Life Organisms - 0401, 0402, 0403, 0404, 0405, 0406
2	0100 Series	Biomedical - 0101, 0102, 0109, 0110, 0111, 0112, 0113, 0114, 0118, 0115, 0116, 0119, 0120.
4	0100 Series	Biomedical - 0103, 0104, 0105, 0106, 0107, 0108, 0117.
4	0200 Series	Behavioral - 0201, 0202, 0203

5.2.3.3 Scheduling

The flight operations and experiment scheduling assumptions utilized in scheduling Flights 523 and 229 are as follows:

a. Experiment Scheduling - See Scheduling Considerations (Appendix D).

b. Flight Operation Scheduling

(1) Three IMU alignments during a 24-hour period (General Electric).

(2) Fourteen hours off duty per astronaut per day consisting of 6 hours sleep, 5 hours rest, 3 hours eating, and personal hygiene. Ten hours on duty (see 24, 25, 26).

(3) System status checks for flight operations require 4.75 man-hours during a 24-hour period (AMPTF adapted, 24, 25, 26). Any system checks required for experiments are included in the experiment setup times.

Communication times for flight operations were not programmed on this flight because the communications contact points were not known at the time of scheduling.

The actual schedules for the first 14 days of Flight 523 are presented in Figure 5-9. This 14-day sample presents the pattern of experiment sequences that can be expected for a complete 45 days. Figure 5-10 shows the schedule for the first three days of Flight 229. Tables 5-9 and 5-10 provide the total amount of time scheduled for each of the experiments, the number of experiment sequences actually scheduled, and the number recommended (31, 32, 33) for a 45 day flight.

This time utilization table, Table 5-8, shows the percentage of time actually scheduled for the four major crew activities and presents the total man-hours of experiments scheduled. As indicated by this chart, approximately 45 percent of the recommended experiment time was actually scheduled for the flight.

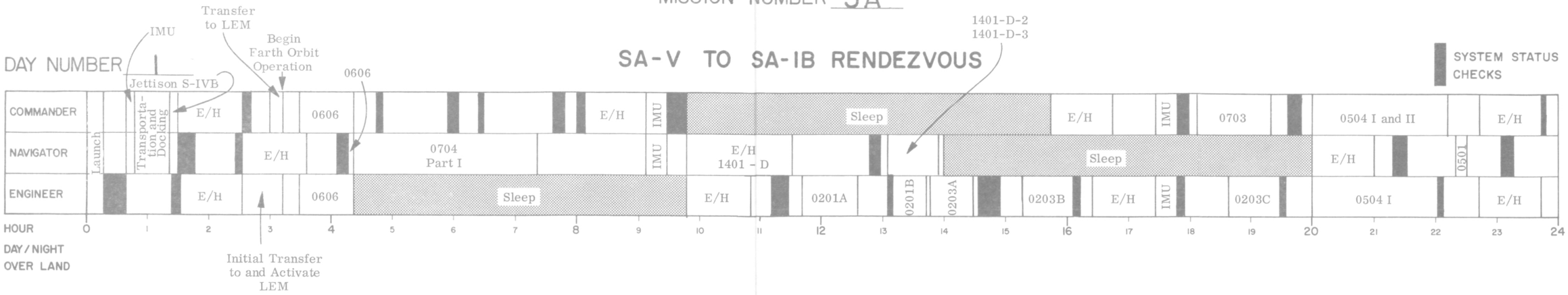
Table 5-8
Time Utilization

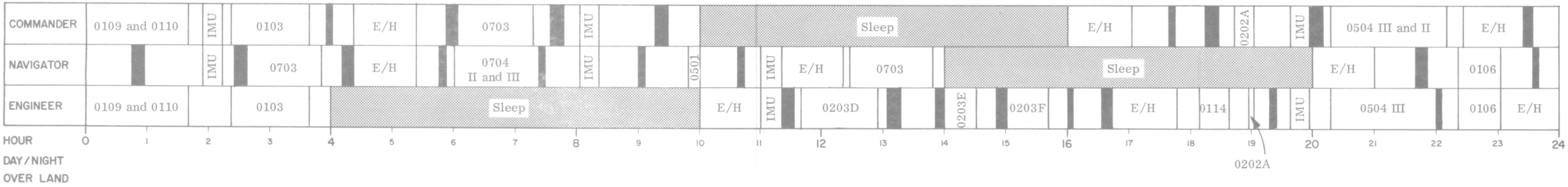
Mission	Time
Experiments	23.7 percent
Flight Operations	8.2 percent
Personal Time	58.4 percent
Unscheduled Time	9.7 percent
Experiment Time Recommended*	1699 man-hours
Experiment Time Scheduled	760 man-hours

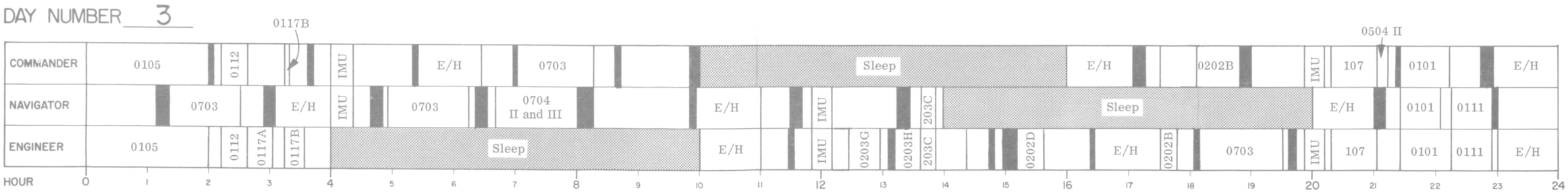
*Grumman number for experiment sequences and subjects tested (see 31, 32, 33), experiment times from Appendix D.

MISSION NUMBER 5A

SA-V TO SA-IB RENDEZVOUS







NOTE: All Blank Areas of 15 Minutes Or More Are Rest Periods.

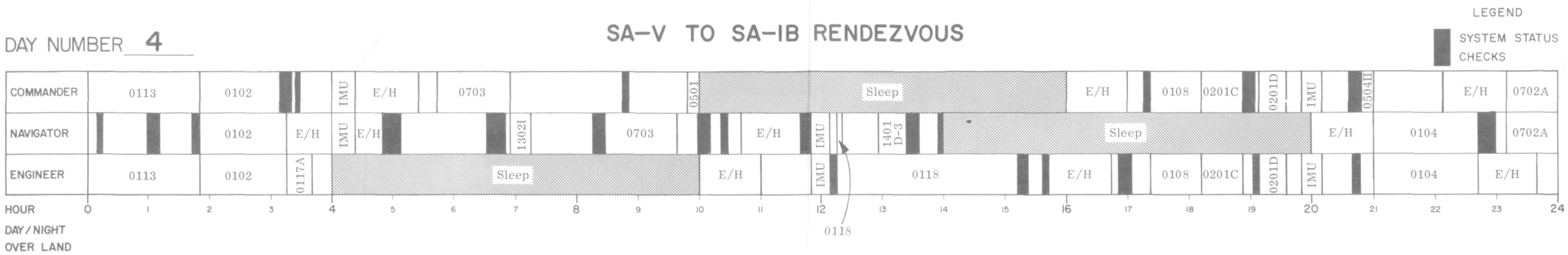
Figure 5-9. Crew Schedule Flight A (Sheet 1 of 5)

5-65-1

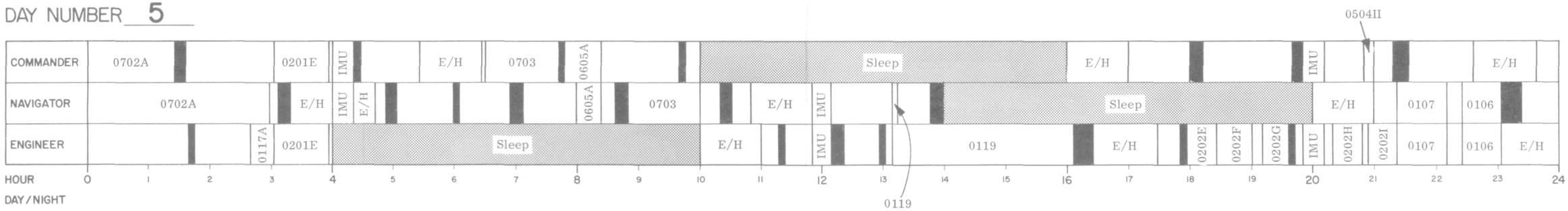
MISSION NUMBER 5A

SA-V TO SA-IB RENDEZVOUS

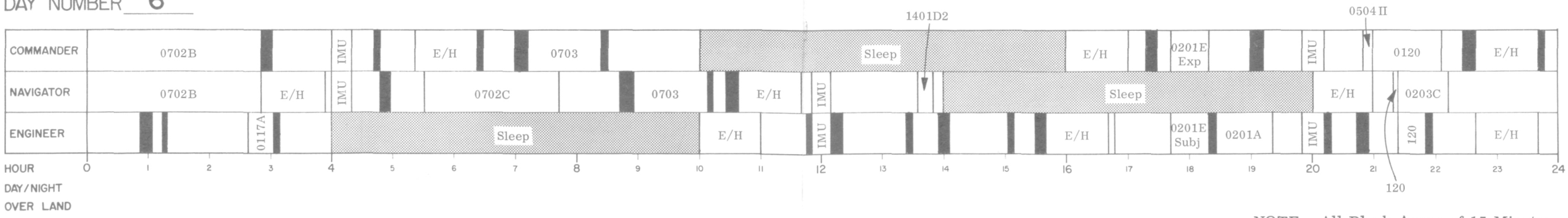
DAY NUMBER 4



DAY NUMBER 5



DAY NUMBER 6



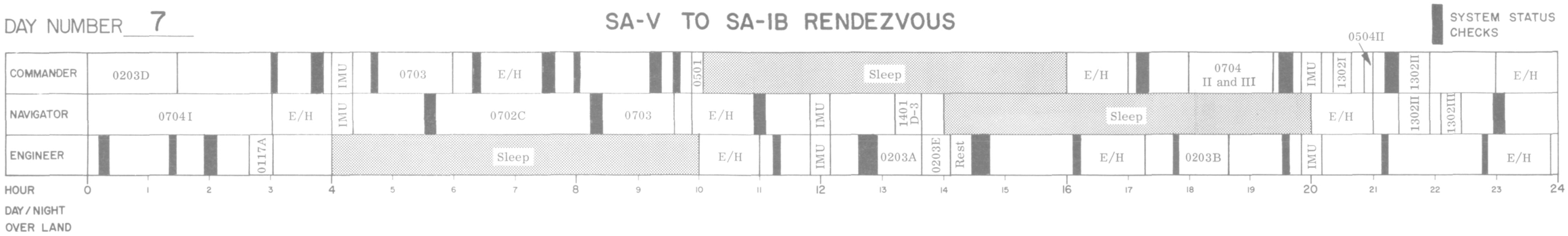
NOTE: All Blank Areas of 15 Minutes Or More Are Rest Periods.

Figure 5-9. Crew Schedule Flight A (Sheet 2 of 5)

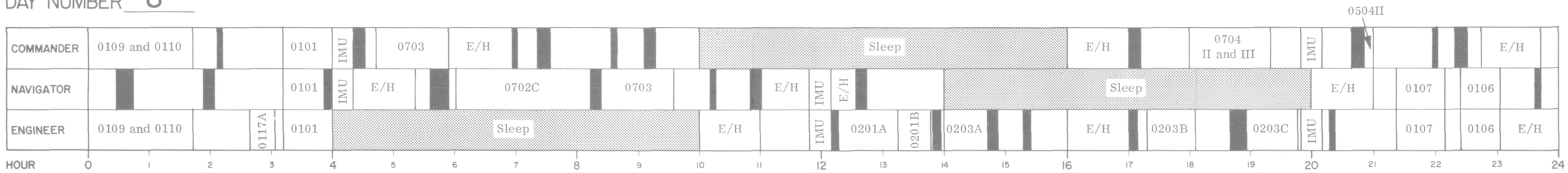
MISSION NUMBER 5A

DAY NUMBER 7

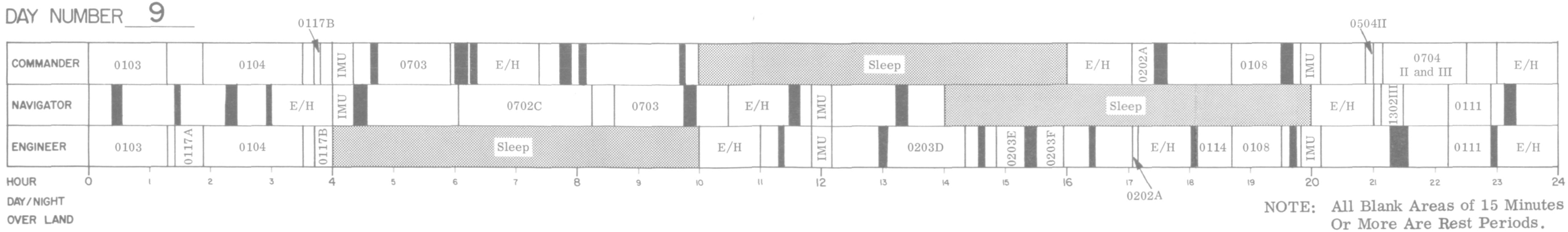
SA-V TO SA-IB RENDEZVOUS



DAY NUMBER 8



DAY NUMBER 9



NOTE: All Blank Areas of 15 Minutes Or More Are Rest Periods.

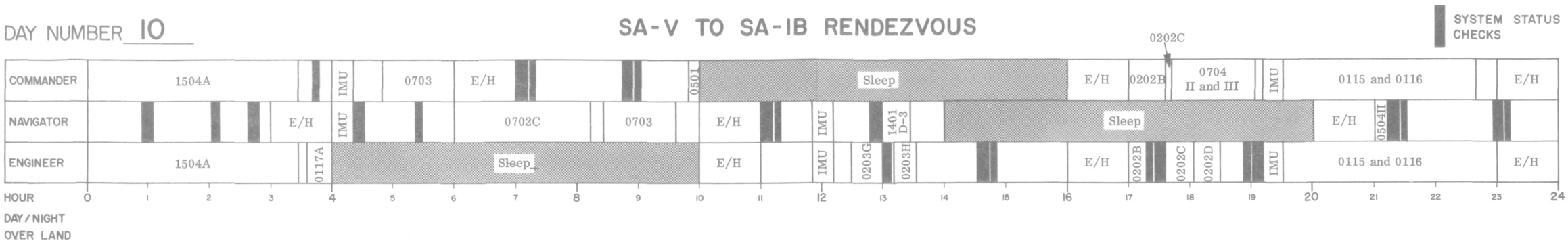
Figure 5-9. Crew Schedule Flight A (Sheet 3 of 5)

5-67.1

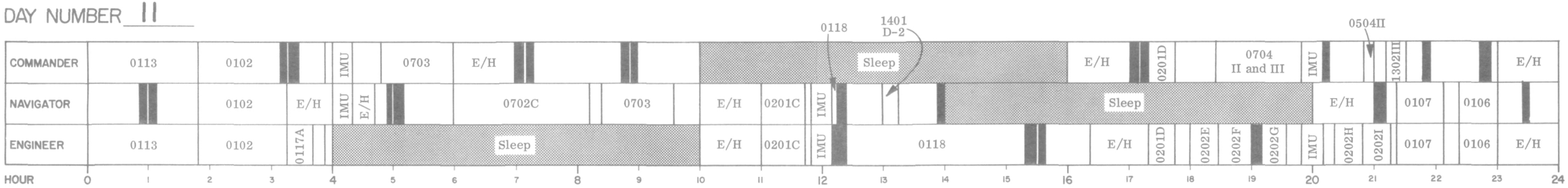
MISSION NUMBER 5A

DAY NUMBER 10

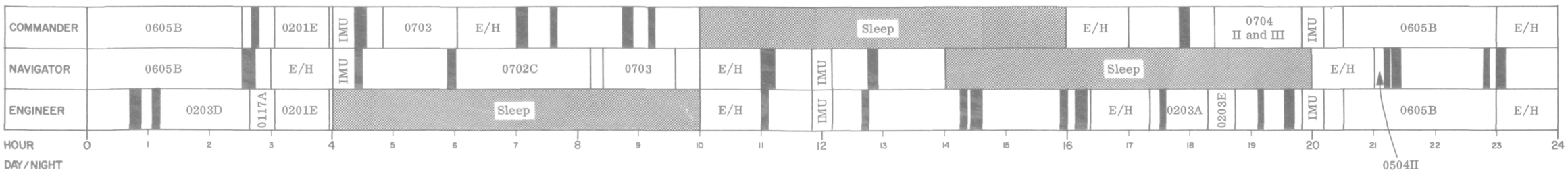
SA-V TO SA-IB RENDEZVOUS



DAY NUMBER 11



DAY NUMBER 12



NOTE: All Blank Areas of 15 Minutes Or More Are Rest Periods.

Figure 5-9. Crew Schedule Flight A (Sheet 4 of 5)

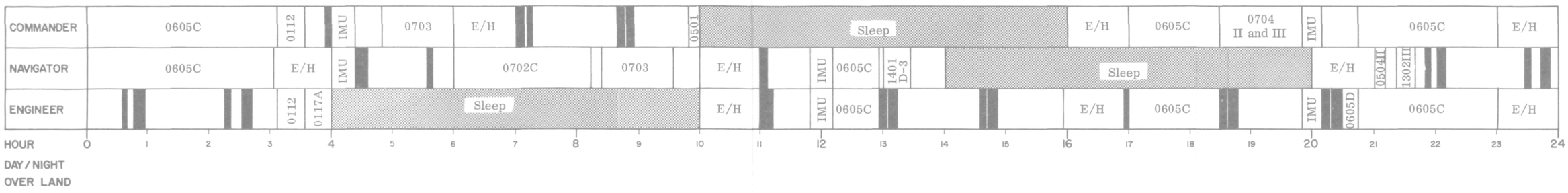
5-68-2

MISSION NUMBER 5A

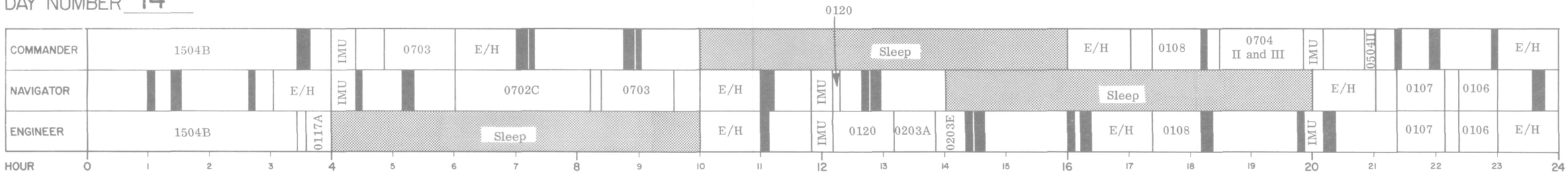
DAY NUMBER 13

SA-V TO SA-IB RENDEZVOUS

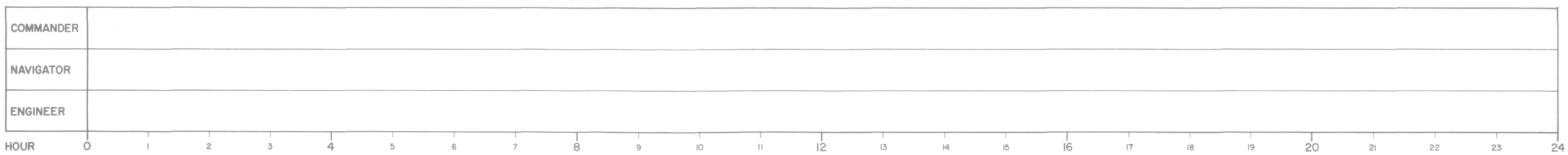
SYSTEM STATUS CHECKS



DAY NUMBER 14



DAY NUMBER _____



NOTE: All Blank Areas of 15 Minutes Or More Are Rest Periods.

Figure 5-9. Crew Schedule Flight A (Sheet 5 of 5)

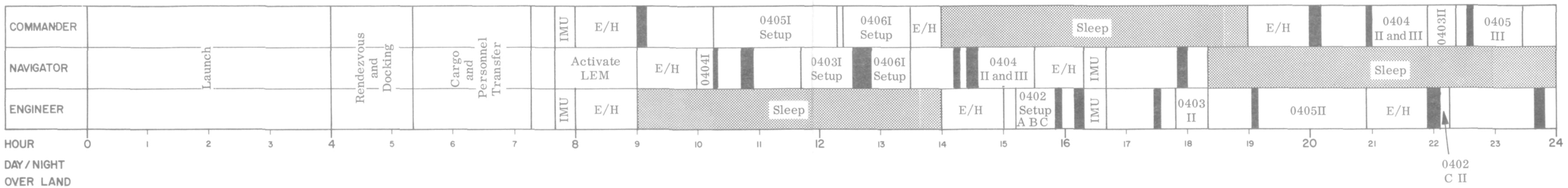
5-69-1

MISSION NUMBER 5B

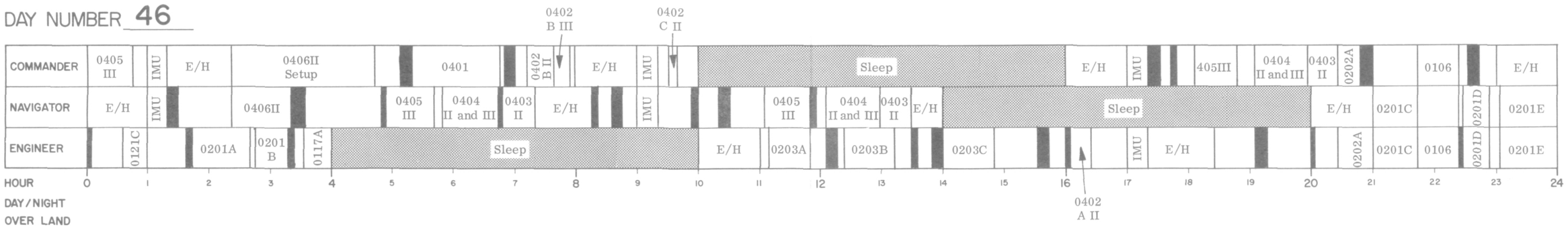
DAY NUMBER 45

SA-V TO SA-IB RENDEZVOUS

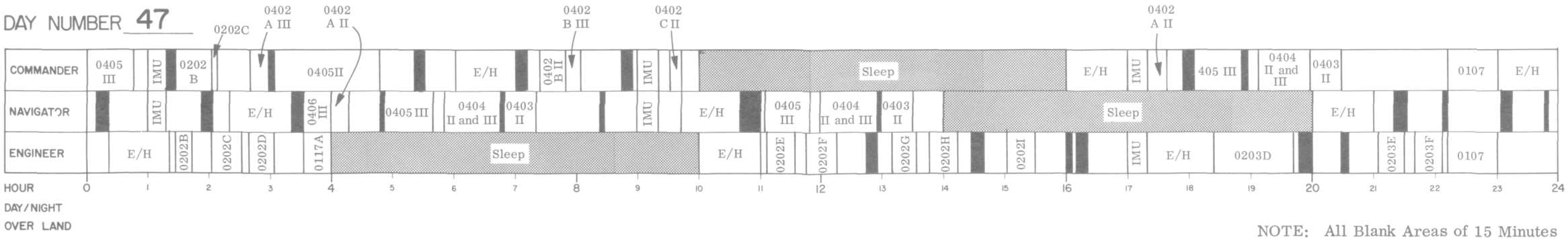
SYSTEM STATUS CHECKS



DAY NUMBER 46



DAY NUMBER 47



NOTE: All Blank Areas of 15 Minutes Or More Are Rest Periods.

Figure 5-10. Crew Schedule Flight B

5-70-2

Table 5-9

Flight No. 523
Summary of Scheduled Experiment Times and Sequences
(14-Day Mission Sample)

Experiment Number	Man-Minutes Expended (14 Days)	Total Number of Sequences	
		Actual*	Recommended**
0101	266	2	15
0102	496	2	15
0103	292	2	12
0104	600	3	12
0105	234	1	6
0106	468	6	17
0107	552	6	17
0108	384	4	9
0109/0110	202	1	15
0111	164	2	9
0112	92	2	9
0113	398	2	5
0114	54	2	36
0115/0116	240	1	4
0115	-	0	18
0117A	250	10	45
0117B	70	2	6
0118	252	4	5
0119	380	2	5
0120	166	2	9
0121	-	0	2
0201A	162	3	15
0201B	96	3	7
0201C	114	3	15
0201D	63	3	7
0201E	114	3	7
0202A	42	2	15
0202B	64	2	15
0202C	46	2	15
0202D	72	3	15

Table 5-9

Flight No. 523
Summary of Scheduled Experiment Times and Sequences (Cont.)
(14-Day Mission Sample)

Experiment Number	Man-Minutes Expended (14 Days)	Total Number of Sequences	
		Actual*	Recommended**
0202E	84	3	15
0202F	96	3	15
0202G	72	3	15
0202H	84	3	15
0202I	84	3	15
0203A	205	5	15
0203B	144	3	15
0203C	98	2	15
0203D	258	3	7
0203E	135	5	7
0203F	56	2	15
0203G	62	2	15
0203H	42	2	15
0501	100	10	45
0504-I	120	1	44
0504-II	140	14	44
0504-III	200	2	4
0504-IV	-	0	1
0601A	-	0	1
0601B	-	0	1
0602A	-	0	5
0602B	-	0	5
0603A	-	0	10
0603B	-	0	3
0604A	-	0	1
0604B	-	0	2
0604C	-	0	1
0605A	33	1	1
0605B	60	1	20
0605C	360	4	20

Table 5-9

Flight No. 523
Summary of Scheduled Experiment Times and Sequences (Cont.)
(14-Day Mission Sample)

Experiment Number	Man-Minutes Expended (14 Days)	Total Number of Sequences	
		Actual*	Recommended**
0605D	-	1	1
0606	171	1	1
0701	-	0	30
0702A	352	1	1
0702B	342	1	1
0702C	1170	9	40
0703	1820	26	90
0704-I	190	1	1
0704-II, III	656	8	40
0705-I	-	0	1
0705-II	-	0	37
1302-I, II	79	1	1
1302-III	80	4	16
1401D-2	45	3	8
1401D-3	130	5	14
1401D-4	-	0	1
1504	404	1	6

*Actual number of sequences scheduled for 14 days (General Electric).
**Recommended number of sequences for 45 days (31, 32, 33).

Table 5-10

Flight No. 229
Summary of Scheduled Experiment Times and Sequences
(3-Day Mission Sample)

Experiment Number	Man-Minutes Expended (3 Days)	Total Number of Sequences	
		Actual*	Recommended**
0101	-	0	15
0102	-	0	15
0103	-	0	12
0104	-	0	12
0105	-	0	6
0106	78	1	17
0107	92	1	17
0108	-	0	9
0109/0110	-	0	15
0111	-	0	9
0112	-	0	9
0113	-	0	5
0114	-	0	36
0115/0116	-	0	4
0115	-	0	18
0117A	50	2	45
0117B	-	0	6
0118	-	0	5
0119	-	0	5
0120	-	0	9
0121A	-	0	2
0121B	-	0	1
0121C	23	1	1
0201A	54	1	15
0201B	32	1	7
0201C	76	1	15
0201D	42	1	7
0201E	76	1	7
0202A	42	1	15
0202B	64	1	15

Table 5-10

Flight No. 229
Summary of Scheduled Experiment Times and Sequences (Cont.)
(3-Day Mission Sample)

Experiment Number	Man-Minutes Expended (3 Days)	Total Number of Sequences	
		Actual*	Recommended**
0202C	46	1	15
0202D	24	1	15
0202E	28	1	15
0202F	32	1	15
0202G	24	1	15
0202H	28	1	15
0202I	28	1	15
0203A	41	1	15
0203B	48	1	15
0203C	49	1	15
0203D	86	1	7
0203E	27	1	7
0203F	28	1	15
0203G	31	0	15
0203H	21	0	15
0401	-	0	6
0402A-I	8	1	1
0402A-II	51	3	80
0402A-III	16	1	40
0402B-I	15	1	1
0402B-II,III	80	2	40
0402C-I	15	1	1
0402C-II	27	3	40
0402C-III	-	0	1
0403-I	50	1	1
0403-II	224	8	135
0403-III	-	0	1
0404-I	15	1	1
0404-II,III	416	8	135
0404-IV	-	0	1

Table 5-10

Flight No. 229
Summary of Scheduled Experiment Times and Sequences (Cont.)
(3-Day Mission Sample)

Experiment Number	Man-Minutes Expended (3 Days)	Total Number of Sequences	
		Actual*	Recommended**
0405-I	120	1	1
0405-II	204	2	45
0405-III	405	9	180
0405-IV	-	0	1
0406-I, II	234	1	1
0406-II	25	1	4
0406-III	-	0	4

*Actual number of sequences scheduled for 3 days (General Electric).
**Recommended number of sequences for 45 days (24, 25, 26).

5.2.4 DATA FLOW

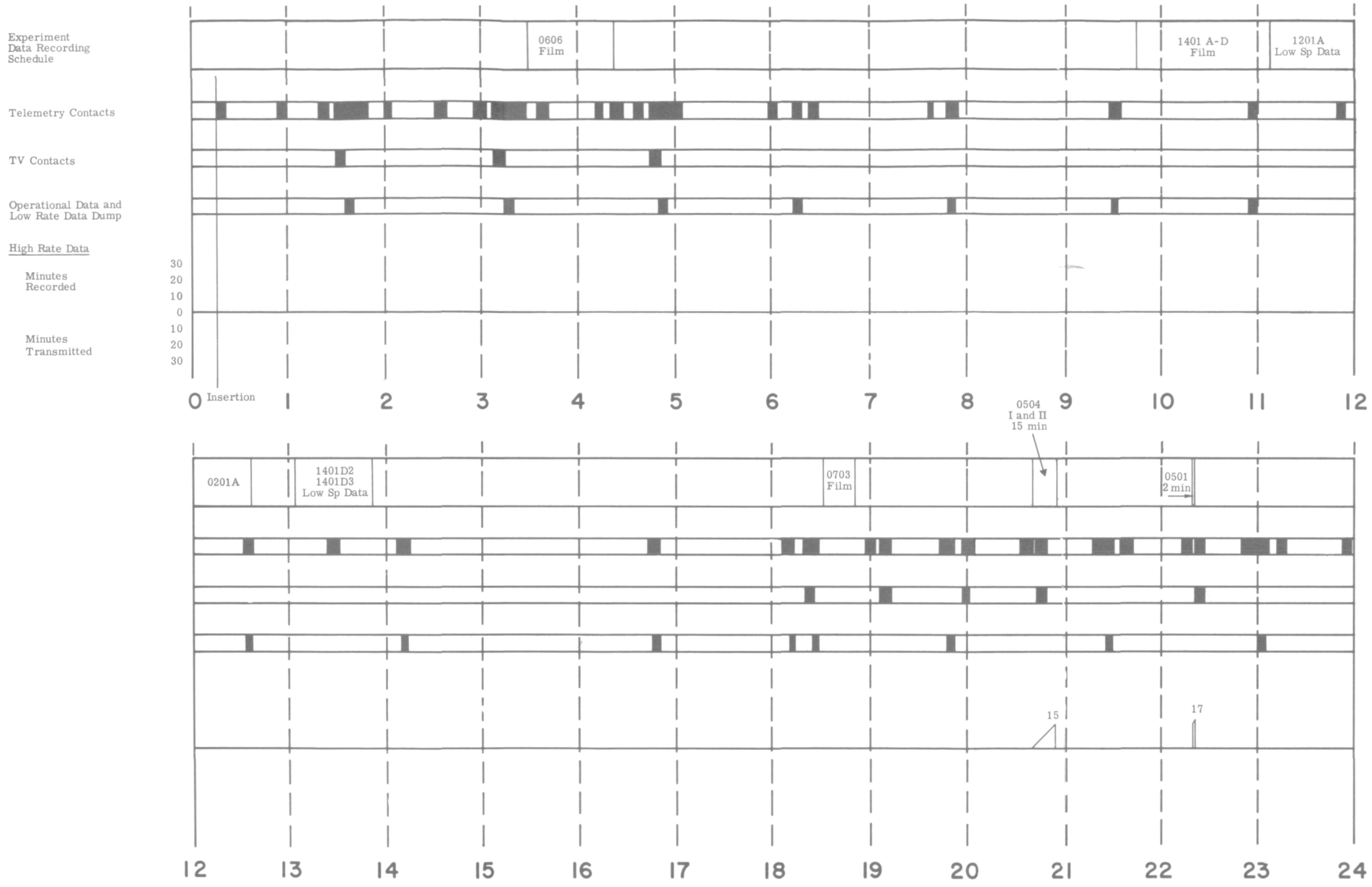
5.2.4.1 Data Recording and Transmission

The data flow was profiled for Mission 5 using the analysis presented in Part II, Section 3, of this report. Instructions for using the profiles are included in the section under the applicable heading.

A data recording and transmission profile for the first three days of each flight of Mission 5 is presented in Figures 5-11 and 5-12. A sample profile was developed for each mission to illustrate graphically the distribution of recording and transmission times.

The top line shows the distribution of experiment data collection sequences and their duration in time. The second line shows the distribution of MSFN S-band telemetry ground station coverages and their duration. The third line shows the coverage distribution of those telemetry stations assumed to have television receiving capability (Goldstone, Madrid, and Canberra). The fourth line shows the distribution of station contacts tentatively assigned for the reception of dumped operational data (and low-sampling-rate experimental data) for the CSM recorder.

MISSION 5A DAY 1

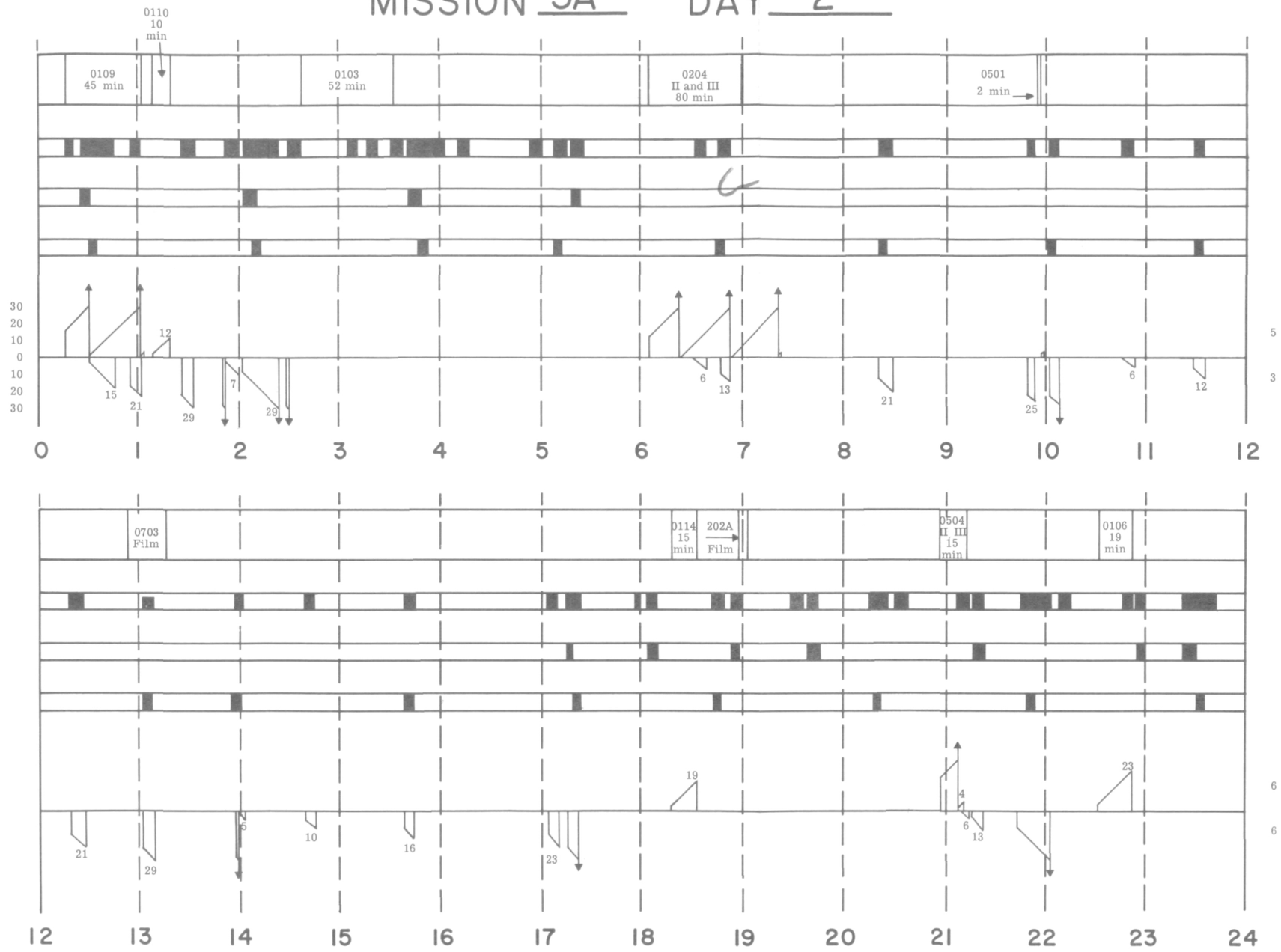


DATA RECORDING AND TRANSMISSION PROFILE

Figure 5-11. Data Recording and Transmission Profile Flight 523 (Sheet 1 of 3)

5-77-1

Part I

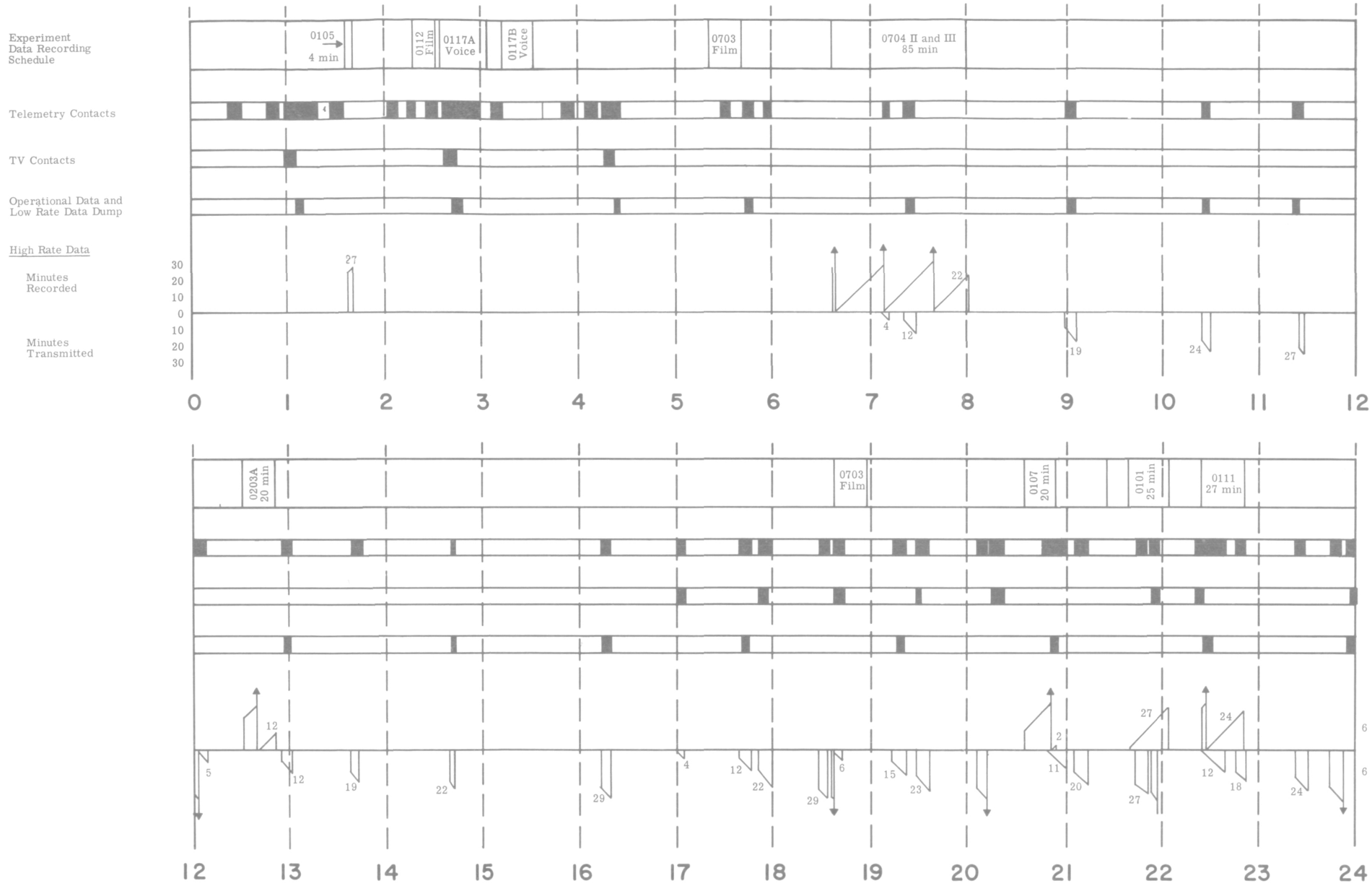


DATA RECORDING AND TRANSMISSION PROFILE

Figure 5-11. Data Recording and Transmission Profile Flight 523 (Sheet 2 of 3)

5-18-2

MISSION 5A DAY 3

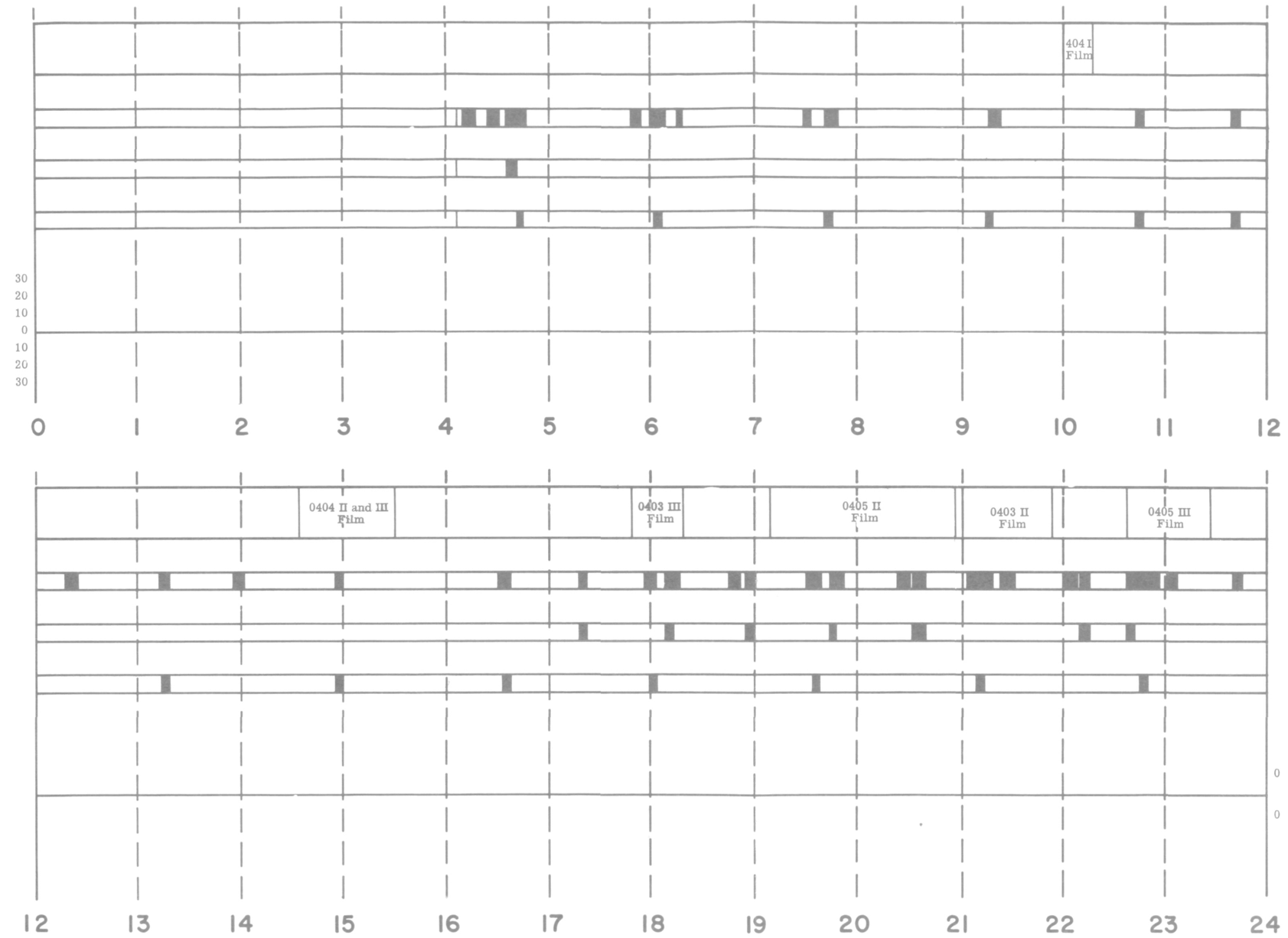


DATA RECORDING AND TRANSMISSION PROFILE

Figure 5-11. Data Recording and Transmission Profile Flight 523 (Sheet 3 of 3)

Part I

MISSION 5B DAY 1



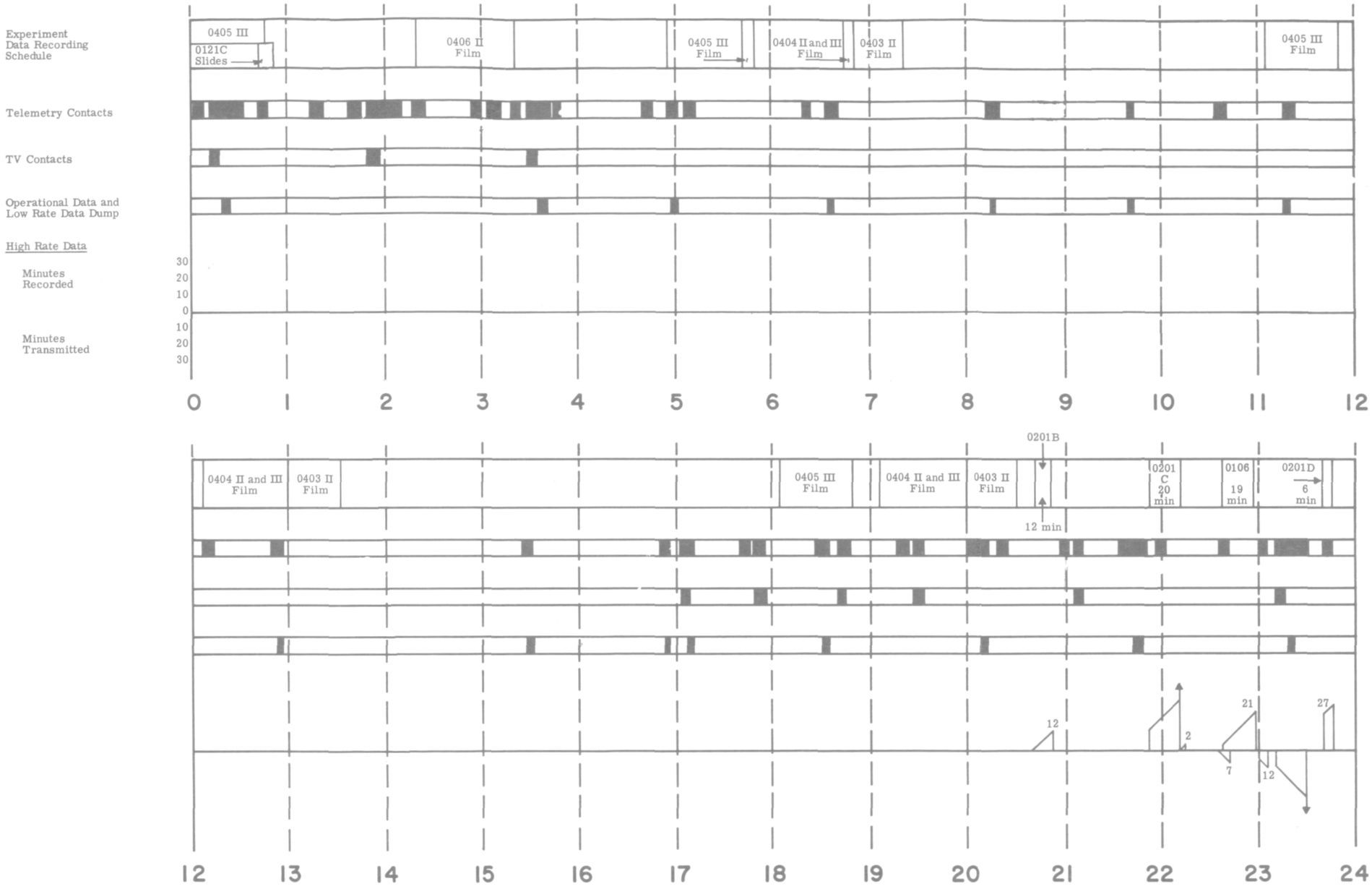
* Television contacts vary from 4 - 8 minutes.
**Operational data contacts are a standard 4 minutes.

DATA RECORDING AND TRANSMISSION PROFILE

Figure 5-12. Data Recording and Transmission Profile Flight 229 (Sheet 1 of 3)

5-80-2

MISSION 5B DAY 2



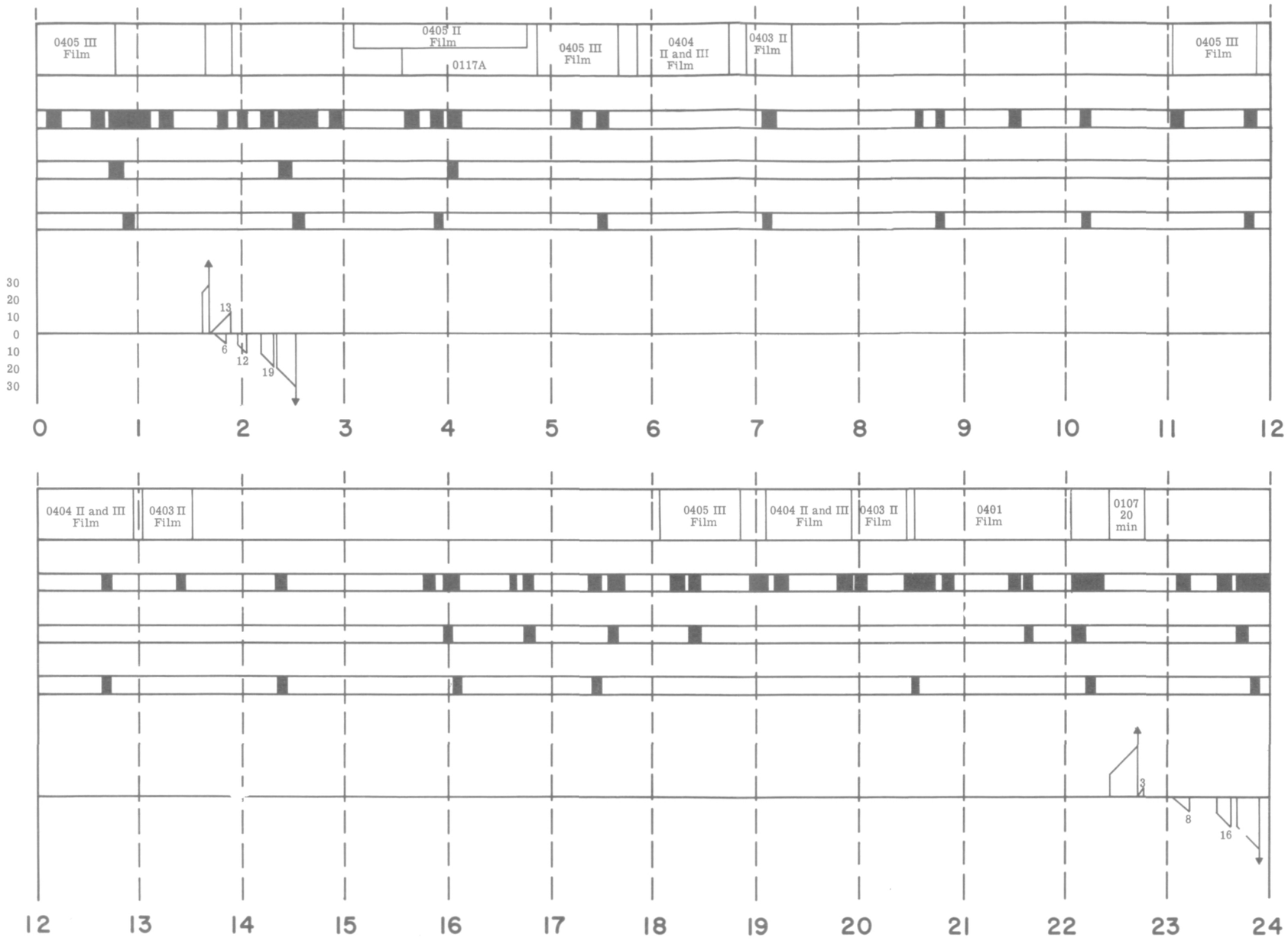
DATA RECORDING AND TRANSMISSION PROFILE

Figure 5-12. Data Recording and Transmission Profile Flight 229 (Sheet 2 of 3)

5-81-1

Part I

MISSION 5B DAY 3



DATA RECORDING AND TRANSMISSION PROFILE

Figure 5-12. Data Recording and Transmission Profile Flight 229 (Sheet 3 of 3)

5-82-2

NOTE

These contacts were assigned on a one-dump-per-orbit basis with no interval greater than 120 minutes whenever possible. When intervals greater than 120 minutes occurred, an extra operational contact was tentatively assigned to allow the dumping of the extra reel of stored operational data tape. Whenever a choice was available, the operational contact was made over the United States.

The fifth line is a profile of the rate of accumulation of the magnetic tape being used to record high-rate PCM data in the LEM, compared to the rate at which reels of this tape may be dumped to the ground telemetry stations. Each upward-pointing arrow indicates the accumulation of one full reel of tape (30 minutes recording time) and each downward-pointing arrow indicates the completed dump of one full reel of tape (30 minutes dumping time).

Note that the profiles for both flights of Mission 5 show a great amount of recording activity during this period, well within the presently planned spacecraft data transmission capabilities.

A noncoverage interval greater than 120 minutes can be expected to occur on the average of once every three orbits with the presently planned network of 14 S-band telemetry ground stations.

5.2.4.2 Magnetic Tape Storage Profile

For each applicable flight of the mission, a sample magnetic tape storage profile was developed for the first three days of the flight to show the nature of the net accumulation of reels of recorded magnetic tape. This profile is derived directly from the fifth line on the data recording and transmission profile. Figures 5-13 and 5-14 are the tape storage profiles for Mission 5, from which no daily accumulation of recorded data is evident.

5.2.4.3 Data Collection Summary

The data collection requirements for Mission 5 are summarized in Tables 5-11 and 5-12.

The first column designates each experiment by experiment number (see 38, 39, 40, 77). The exception to this is series 0100, the biomedical experiments, and series 0200, behavioral experiments. The next column is the maximum number of special analog channels that must be recorded at any one time because of that experiment. The next

Table 5-11

Data Collection Summary

Mission 5A

Experiment	Data Recording							Recovery Requirements				Remarks
	Max. No. of Channels in Use at One Time		Minutes per Sequence	Total No. of Sequences	Total Minutes Recorded	Transmission Time Avail. (Minutes)	Recorded Back Log (Minutes)	Magnetic Tape		Pounds of Film	Miscellaneous	
	Anal.	Dig.						Reels	Lbs.			
	*	*				**	***	**	**			* Assumed time sharing of channels as required. ** Summed only for total mission ***Difference - Recorded minus Transmitted - only for total mission.
0100	6	1	*	111**	3904					X	X	* Varies for each experiment in series. See Data Collection Charts, Part II, Section 3. ** Varies for each experiment in series, number (111) is total for series for which recorder is used.
0200	1	1	*	18**	1410					145	X	* Varies for each experiment in series. See Data Collection Charts. ** Varies for each experiment in series; number (18) is total for series for which recorder is used. X Indicates a requirement - no data on rates.

Table 5-11

Data Collection Summary (Cont.)

Mission 5A

Experiment	Data Recording							Recovery Requirements				Remarks
	Max. No. of Channels in Use at One Time		Minutes per Sequence	Total No. of Sequences	Total Minutes Recorded	Transmission Time Avail. (Minutes)	Recorded Back Log (Minutes)	Magnetic Tape		Pounds of Film	Miscellaneous	
	Anal.	Dig.						Reels	Lbs.			
0501		1	2	40	80					X		
0504		1	15	4 at 5 44 at 10	460						X*	* Collector panels
0601		1	330	1	330					7*	X**	* 2800 feet of 16 mm film **One picture for each static test (10 total)
0602A	1	1	30	5	150					10		
B										X*		* 4000 feet of 16 mm film
0603A		1	55	10	550					X*		
B		1	30	3	90					10		* 200 to 400 feet of film per active orbit
0604		1	40	2	80					X		
0605			70	20	1400					X	X*	* Two pounds of samples plus 10 photos
0606			**		**						30*	* Photo emulsion package **10 to 14 days
												X Indicates a requirement - no data on rates.

Mission 5A

Table 5-11

Data Collection Summary (Cont.)

Experiment	Data Recording							Recovery Requirements				Remarks
	Max. No. of Channels in Use at One Time		Minutes per Sequence	Total No. of Sequences	Total Minutes Recorded	Transmission Time Avail. (Minutes)	Recorded Back Log (Minutes)	Magnetic Tape		Pounds of Film	Miscellaneous	
	Anal.	Dig.						Reels	Lbs.			
0701	1		30	30	900							
0702	1	1	45	40	1800					20*	100**	* 1000 feet of film **1600 - 4 x 5 inch glass plate photographs
0703		1	20	90	1800							
0704		1	125	40	5000							
0705A	1	1	90	37	3330							
1302										4		
1401	1	1	*								X	* Included in normal personal hygiene time estimated equivalent to 100 feet of single channel tape per day.
1504	1	1	340	2	680					X*		* Five rolls - 8 or 16 mm film
Total	6**	1**			13904*	15967	0	0	0	191+	130+	* Time for 0504, 0601, 0603, 0604 not included in that data is low rate, 0605 and 0606 not collected on magnetic tape. **Total number of channels in use at one time during mission, not a column summation. X Indicates a requirement - no data on rates.

Table 5-12

Data Collection Summary

Mission 5B

Experiment	Data Recording							Recovery Requirements				Remarks
	Max. No. of Channels in Use at One Time		Minutes per Sequence	Total No. of Sequences	Total Minutes Recorded	Transmission Time Avail. (Minutes)	Recorded Back Log (Minutes)	Magnetic Tape		Pounds of Film	Miscellaneous	
	Anal.	Dig.						Reels	Lbs.			
	*	*				**	***	**	**			* Assumed time sharing of channels as required. ** Summed only for total mission. ***Difference - Recorded minus Transmitted - only for total mission.
0100	6	1	*	111**	3904					X	X	* Varies for each experiment in series, see Data Collection Charts, Part II, Section 3. ** Varies for each experiment in series; number (111) is total for series for which recorder is used.
0200	1	1	*	18**	1410					145	X	* Varies for each experiment in series, see Data Collection Charts, Part II, Section 3. ** Varies for each experiment in series; number (18) is total for which recorder is used. X Indicates a requirement - no data on rates.

Table 5-12

Data Collection Summary (Cont.)

Mission 5B

Experiment	Data Recording							Recovery Requirements				Remarks
	Max. No. of Channels in Use at One Time		Minutes per Sequence	Total No. of Sequences	Total Minutes Recorded	Transmission Time Avail. (Minutes)	Recorded Back Log (Minutes)	Magnetic Tape		Pounds of Film	Miscellaneous	
	Anal.	Dig.						Reels	Lbs.			
0501		1	2	40	80					X		
0401			7	6	42					X	X	
0402			4	40	160					X	X	
0403			28	45	1260							
0404			8	45	360					X	X	
0405												
0406			10	6	60							
Total	6**	1**			5394**	15967				145	X	* Time for 0400 series not included in that the data is not collected on magnetic tape. **Total number of channels in use at one time during mission, not a column summation; assumes time sharing as required. X Indicates a requirement - no data on rates.

MAGNETIC TAPE STORAGE PROFILE
MISSION 5A

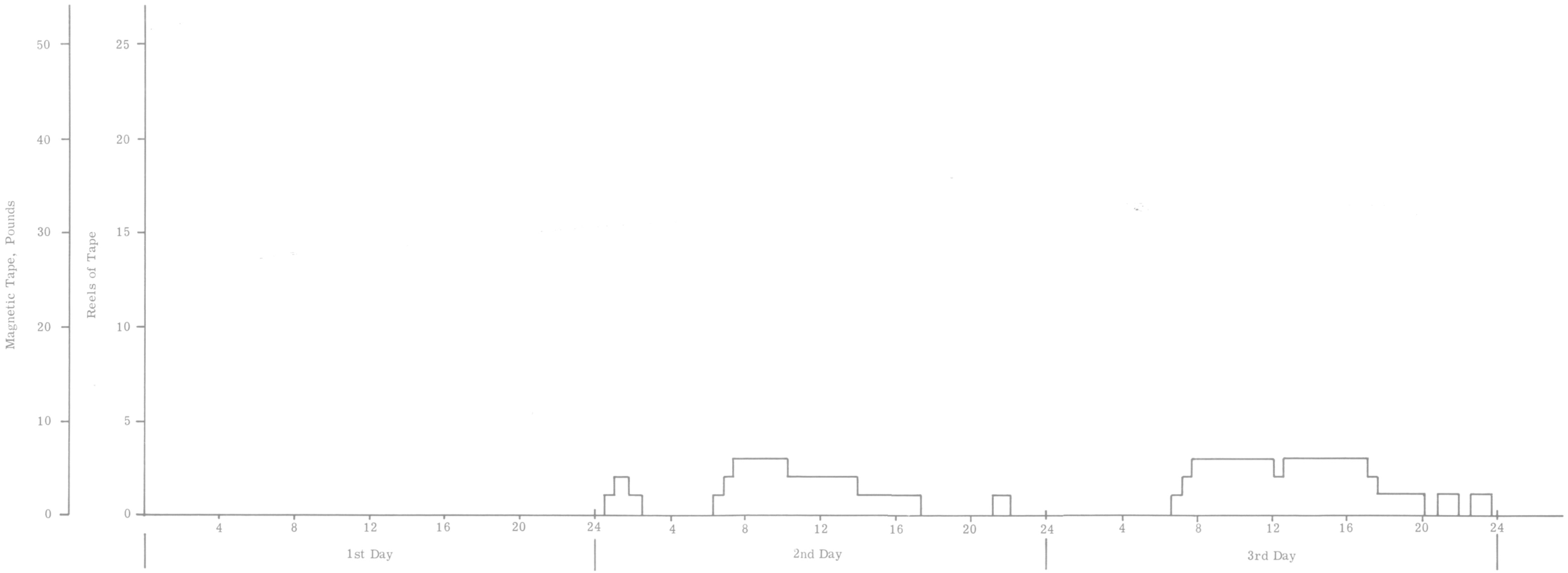


Figure 5-13. Magnetic Tape Profile
Flight 523

5-89-1

MAGNETIC TAPE STORAGE PROFILE
MISSION 5B

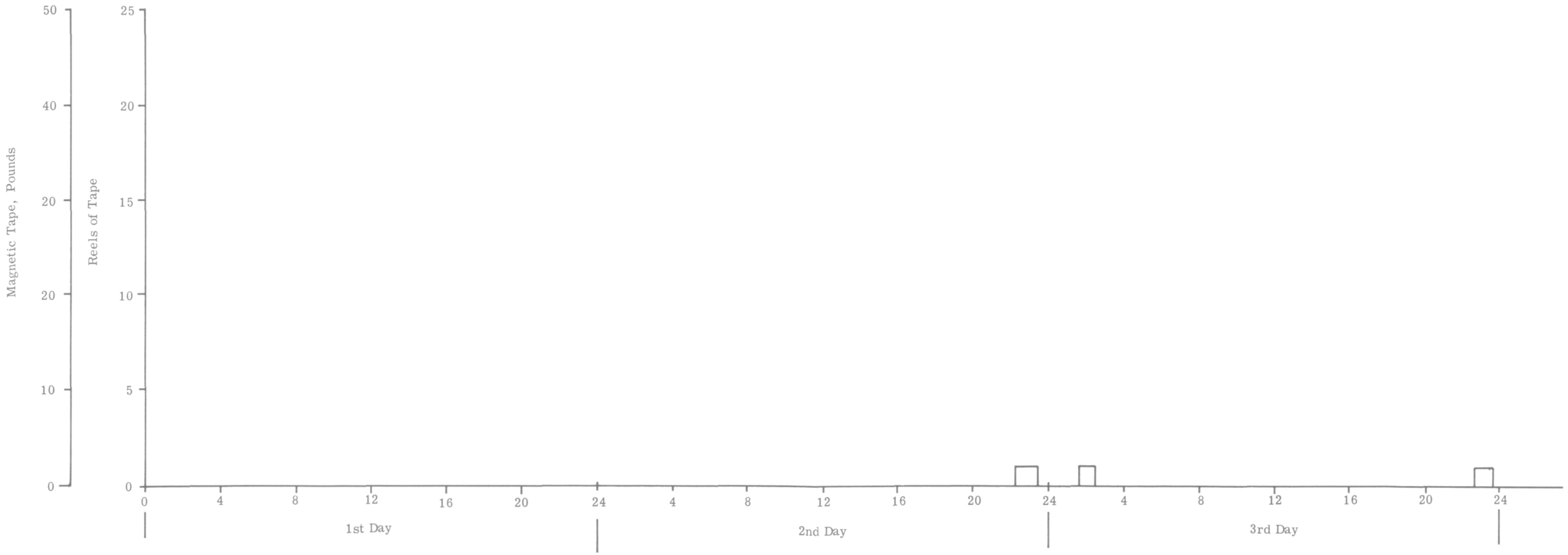


Figure 5-14. Magnetic Tape Profile Flight 229

5-90-2

column indicates whether or not the digital output of a PCM multiplexer must be recorded. The fourth column indicates the minutes of recording time required by each experiment sequence. The fifth and sixth columns indicate the total number of sequences and total minutes of recording time, respectively, required by that experiment during the course of the mission.

The next column contains a graphically derived value of the total transmission time available during the mission and is entered only in the over-all totals. The figure given for "Recorded Backlog" is the difference between the over-all total "high rate" PCM and analog data recording time and the over-all total available transmission time. Further experiment data collection details are given in Part II, Section 3.

Several forms of data are required on Mission 5 for which no detailed estimates of weight and volume are available. For other experiments, the chart shows that both the film and the excess magnetic tape requirements exceed the assumed recoverable payload weight of 250 pounds on Flight A.

5.2.5 TRAJECTORY ANALYSIS

5.2.5.1 Trajectory Description

Mission 5 incorporates Flights 523 and 229 in a Saturn V to Saturn IB rendezvous. The Saturn V booster launches the first vehicle directly into the mission orbit of a 200-n. mi. altitude and an orbital inclination of 29.5 degrees. Vehicle A remains in this orbit until Vehicle B is launched on day 44. Vehicle B is launched by the Saturn IB booster into an 80-n. mi. parking orbit. Vehicle B performs transposition and docking in the parking orbit and then transfers to the 200-n. mi. mission orbit utilizing a Hohmann ellipse. Rendezvous occurs in the mission orbit. All active maneuvers are performed by Vehicle A. After the docking of the two vehicles, the requisite cargo and personnel are transferred, and the vehicles continue to orbit in a docked condition in the mission orbit.

After approximately one day, CSM A separates, leaving LEM/LAB A attached to Vehicle B. The CM A re-enters and is recovered. Vehicle B continues in mission orbit for 44 days, after which time the CM B re-enters and is recovered.

A detailed listing of flight dynamics parameters for all powered phases of the mission is given in Tables 5-13 and 5-14.

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Table 5-13

Mission 5A - Saturn V to Saturn IB Rendezvous
Saturn V Reference Trajectories, Low-Earth Orbits (200-N. Mi. Circular Orbit, i = 28.5 Degrees)

Trajectory Event	Time (seconds)	Geodetic Latitude (degrees)	Longitude (degrees)	Velocity[1] (feet per second)	Altitude (feet)	Flight[1] Path Angle (degrees)	Azimuth[2] (degrees)	Vehicle Weight (pounds)
1. Liftoff	0.00	28.65	-80.64	1,340	0	0.00	---	6,490,004
2. End Vertical Rise, Begin Zero-Lift Flight	12.00	28.65	-80.64	1,343	451	3.39	90.0	6,143,543
3. Shutdown of S-IC Inboard Engine	154.57	28.64	-80.04	7,709	209,792	28.94	90.6	2,027,385
4. Shutdown of S-IC Outboard Engines, Begin Coast	158.57	28.64	-79.97	8,101	224,988	28.53	90.6	1,934,996
5. Jettison of S-IC, S-II Ignition Begin Pitch-Up Maneuver,[3] Mixture Ratio = 5.0	162.37	28.64	-79.90	8,044	239,479	27.85	90.7	1,553,350[4]
6. End Pitch-Up Maneuver, Change Mixture Ratio to 5.4	172.37	28.64	-79.72	8,113	276,251	26.36	90.8	1,529,054
7. Jettison S-IC/S-II Interstage Adapter Section	188.57	28.64	-79.42	8,270	332,984	24.30	91.0	1,475,844[5]
8. Jettison Launch Escape System	193.57	28.64	-79.32	8,325	349,854	23.69	91.1	1,454,237[6]
9. Change Mixture Ratio to 4.7	386.28	28.44	-74.49	12,369	825,033	8.72	94.1	937,476[9]
10. Shutdown of S-II, Begin Coast	534.19	27.93	-68.73	17,454	1,075,177	5.95	97.2	605,388
11. Jettison S-II, S-IVB Ignition	538.99	27.91	-68.50	17,440	1,083,658	5.71	97.4	504,457[7]
12. Burnout S-IVB, End Integrated Trajectory	1,014.32	22.15	-42.97	25,231	1,214,396	0.00	109.7	273,495[8]
13. Service Module Ignition for Deboost Maneuver	0.00			25,231	1,215,220	0.00		21,421
14. Service Module Burnout	12.49	28.5	47.0	24,872	1,215,200	0.00		20,597
15. Command Module Re-Entry	1,607.20	8.0	-145	25,831	400,000	-1.52		10,303

(1) Inertial Quantities
(2) Relative Quantities
(3) Pitch-up of one degree per second for 10 seconds approximates optimum trajectory, after which a pitch-down rate of 0.0798 degrees per second is maintained to S-IVB burnout.
(4) After Jettison Event of 381,645 Pounds
(5) After Jettison Event of 9,770 Pounds
(6) After Jettison Event of 8,200 Pounds
(7) After Jettison Event of 100,921 Pounds
(8) S-IVB jettison weight is 40,539 pounds including 3800 pounds LEM adapter section.
(9) The difference between 8 and 9 represents the payload left in orbit or consumed during stay plus the spent S-IVB.

Note: Events 1 through 12 derived from reference 49, Table 1.
Events 13 through 15, General Electric Apollo Support Department calculations.

Table 5-14

Mission 5B - Saturn V to Saturn IB Rendezvous
Saturn IB Reference Trajectories; 200-N.Mi. Circular Orbit by Hohmann Transfer
(Perigee = 80-N.Mi. Circular Orbit, i = 26.5 Degrees)[10]

Trajectory Event	Time (seconds)	Geodetic Latitude (degrees)	Longitude (degrees)	Velocity[1] (feet per second)	Altitude (feet)	Flight[1] Path Angle (degrees)	Azimuth[2] (degrees)	Vehicle Weight (pounds)
1. Liftoff	0.00	28.65	-80.64	1,340	0	0.00	----	1,298.819
2. End Vertical Rise, Begin Zero-Lift Flight	10.00	28.65	-80.65	1,345	506	4.50	90.0	1,234.545
3. Shutdown of S-IB Inboard Engines	134.72	28.64	-80.11	7,242	176,584	27.36	90.5	434.345
4. Shutdown of S-IB Outboard Engines (Burnout) Begin Coast	141.64	28.64	-80.00	7,623	199,816	26.34	90.6	412.472
5. Jettison of S-IB; S-IVB Ignition; Mixture Ratio = 5.0; Constant Attitude Rate	145.94	28.64	-79.93	7,564	214,082	25.50	90.6	306.646[3]
6. Jettison Ullage Cases and Thermolag, Change Mixture Ratio to 5.5	155.94	28.64	-79.75	7,648	245,726	23.69	90.8	301.599[4]
7. Jettison Launch Escape System	165.94	28.64	-79.58	7,772	275,618	21.96	90.9	287.963[5]
8. Change Mixture Ratio to 4.7	440.00	28.22	-71.48	16,341	577,566	-1.18	95.7	138.981
9. S-IVB Burnout (80-N.Mi. Parking Orbit) End Integrated Trajectory	595.54	27.11	-62.61	25,665	486,007	0.00	100.2	69,775
10. Service Module Ignition for Perigee Maneuver	3:10:00[9]	21.0	-84.1	25,665	80[11]	0.00		34.770[6]
11. Service Module Burnout, Coast to Apogee[7]	3:10:12	21.0	-84.1	25,879	80	0.00		33.968
12. Service Module Ignition for Apogee Maneuver	3:55:00	21.0	+84.42	25,016	200	0.00		33.968
13. Service Module Burnout	3:55:11	21.0	+84.42	25,229	200	0.00		33.184[10]
14. Service Module Ignition for Deboost Maneuver	00:00:00			25,229	200	0.00		21.421[8]
15. Service Module Burnout	00:00:12	-28.5	+47	24,870	200	0.00		20.597
16. Command Module Re-Entry	00:26:47	+8	-145	25,831	66	-1.52		10.303

(1) Inertial Quantities
(2) Relative Quantities
(3) After Jettison Event of 105,826 Pounds
(4) After Jettison Event of 235 Pounds
(5) After Jettison Event of 8,200 Pounds
(6) After Jettison Event of 34,985 Pounds
(7) Each S/M operation provides an additional 10 percent velocity contingency over the velocity increment required.
8. The difference between the weights of events 13 and 14 represents payload left in orbit or expended during stay.
9. Allows 3 hours for crew tasks (hours:minutes:seconds).
10. For inclination of 29.5 degrees, reduction in weight is 90 pounds reference 49, Table 1.
11. Nautical Miles

Notes: Events 1 through 12 derived from reference 49.
Events 13 through 15, General Electric Apollo Support Department calculations.

Data for the launch from liftoff through injection into the 80-n. mi. circular orbit is taken from Reference 49. The remainder of the data is obtained through computation. The launch and re-entry traces for Vehicle B are shown on a uniform grid projection in Figure 5-15. Ground tracking coverage and illumination for the ascent phases is given in Figure 5-16.

The ground traces for the later orbits can be obtained using the uniform grid projection, with ascending nodes plotted, in conjunction with transparent overlays. These materials are included in Volume IV.

5.2.5.2 Assumptions

The following assumptions were used to construct the nominal mission trajectory:

a. Vehicle A is launched at 1200 hours GMT.
b. Vehicle B is constrained to a three-hour launch window.
c. Vehicle B is launched at a nominal time to preclude the necessity for rendezvous maneuvering.

5.2.5.3 Discussion

Off-nominal launch times of Vehicle B require a plane change maneuver and a phasing maneuver to effect rendezvous. The weight margin on Vehicle A (Flight 523) is quite large compared to the margin available on Vehicle B (Flight 229) (see Table 5-2). If needed, 41,000 pounds of SPS fuel can be loaded aboard Vehicle A. Since the only requirement made on the SPS of Vehicle A is the deboost impulse, it is reasonable to assume that Vehicle A provides any active maneuvering required for rendezvous.

The maneuver requirements necessary to affect rendezvous are examined in the supporting study, Rendezvous Analysis, Part II, Section 5. This reference indicates that a three-hour launch window for Vehicle B requires a one-degree plane change capability. This is equivalent to a velocity impulse of 450 feet per second.

The analysis also indicates that only 14.5 minutes of delay in launch can be made up by Vehicle B due to the ground rule which limits Vehicle B to three orbits in the 80-n. mi. parking orbit. Therefore, active maneuvers by Vehicle A are necessary to accomplish phasing and to open the launch window for Vehicle B to three hours.

Two possible modes to accomplish the phasing maneuver are diagramed in Figures 5-17 and 5-18. Mode 1, for late launches of Vehicle B, places Vehicle A into an elliptic

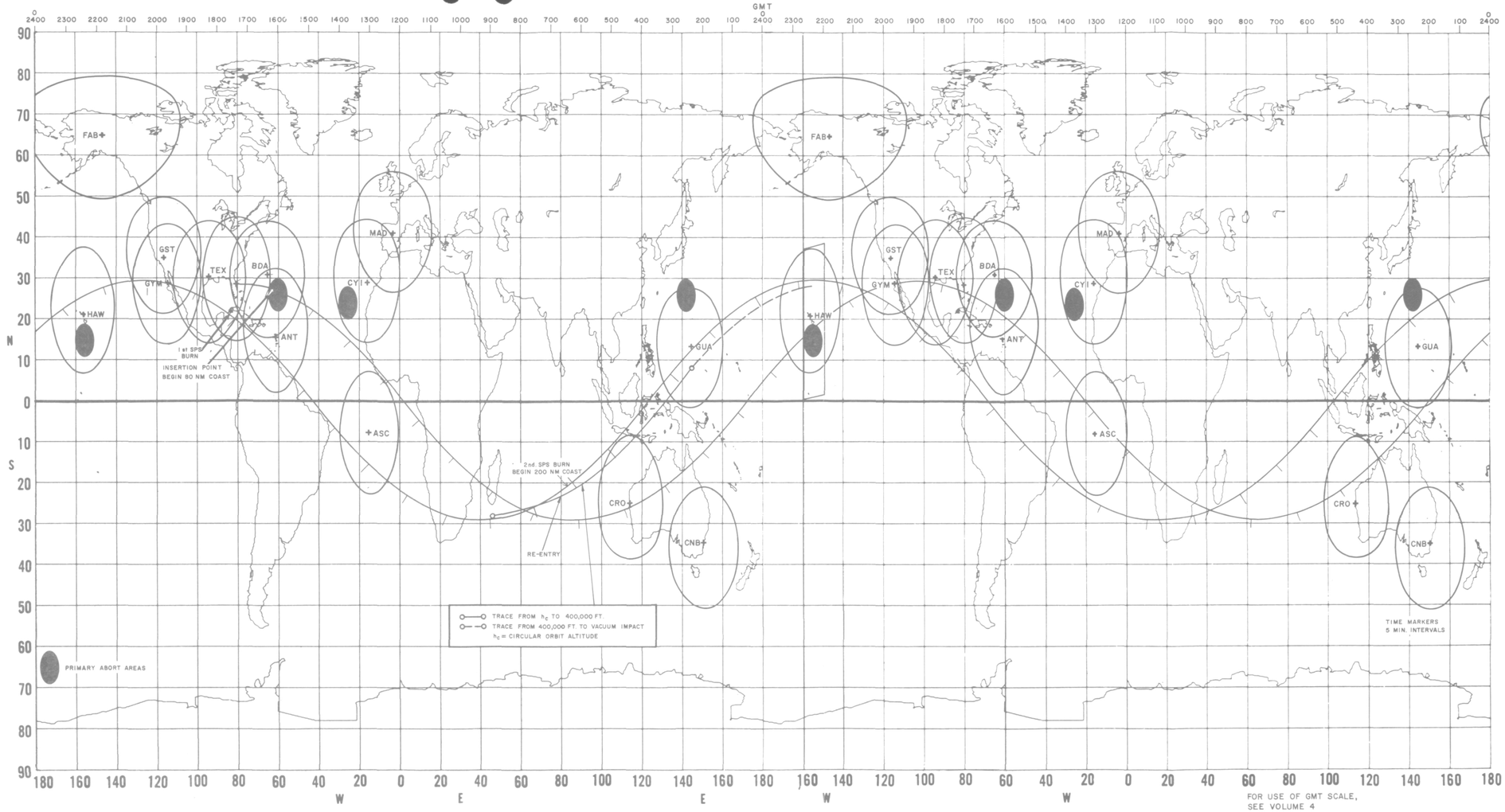

Figure 5-15. Mission 5 - Vehicle B Launch, Insertion, and Re-entry Trajectories

5-95-1

5-95-2

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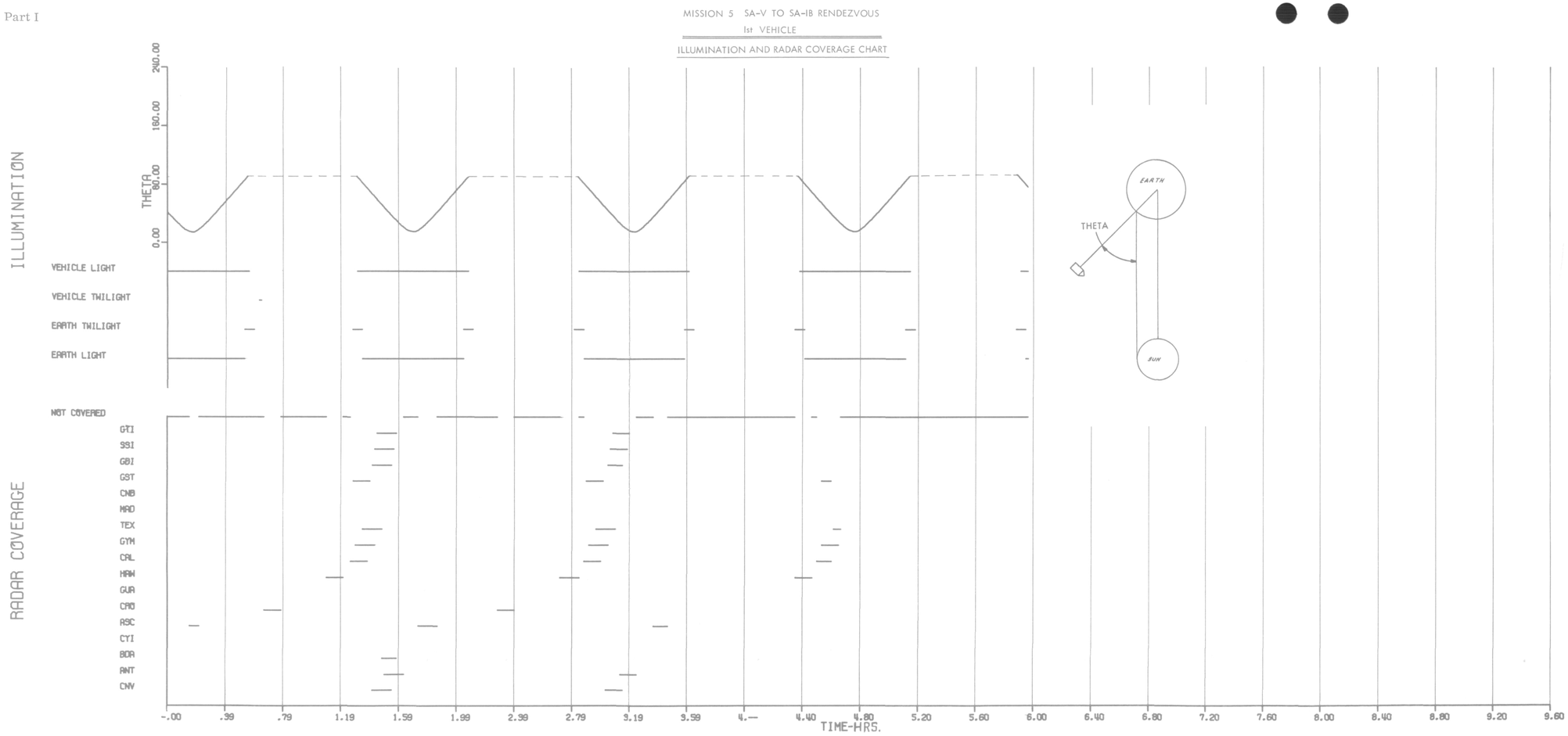

Figure 5-16. Illumination and Radar Coverage

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MISSION 5 RENDEZVOUS MANEUVER (1)

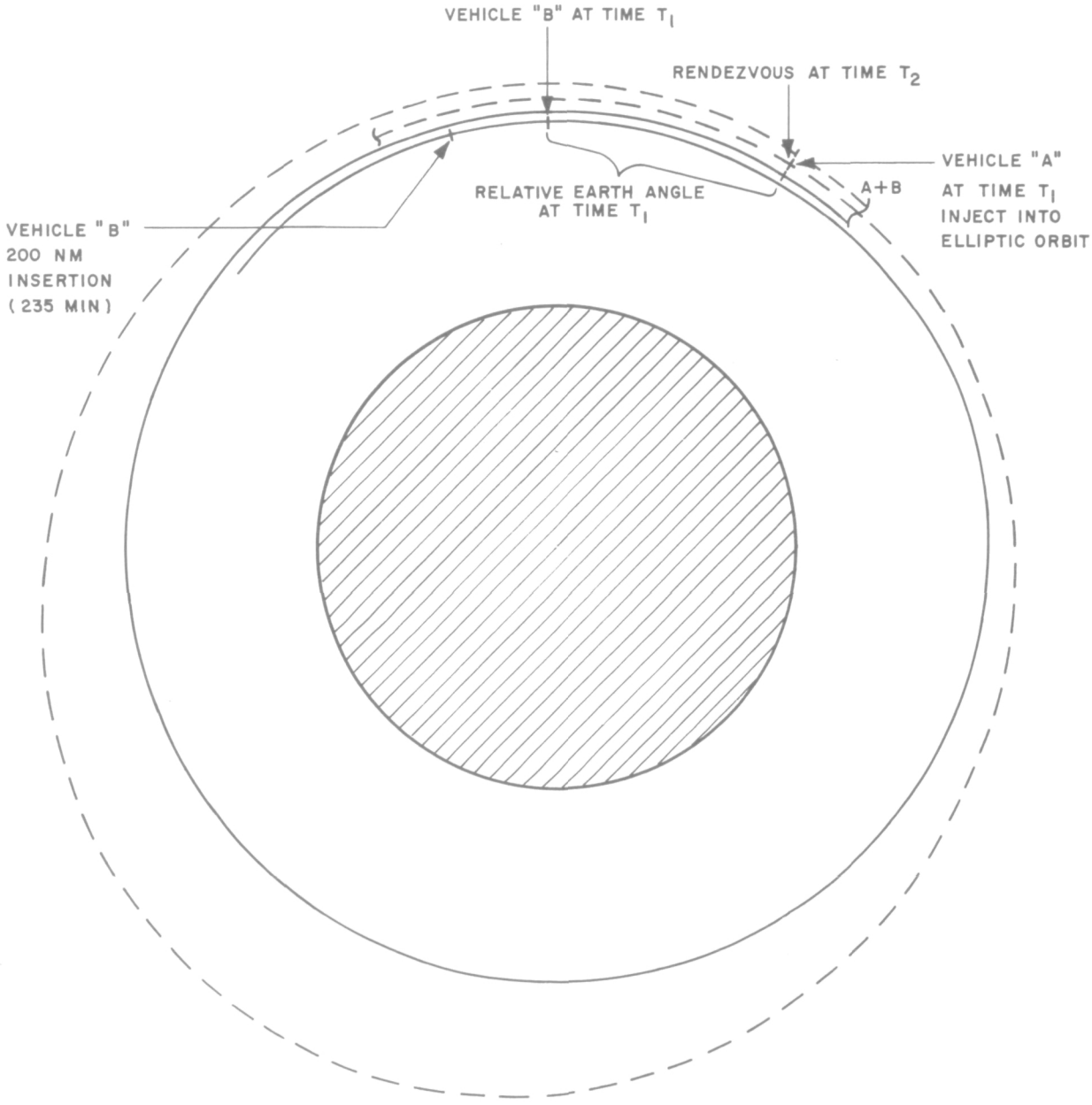

Figure 5-17. Rendezvous Maneuver (1)

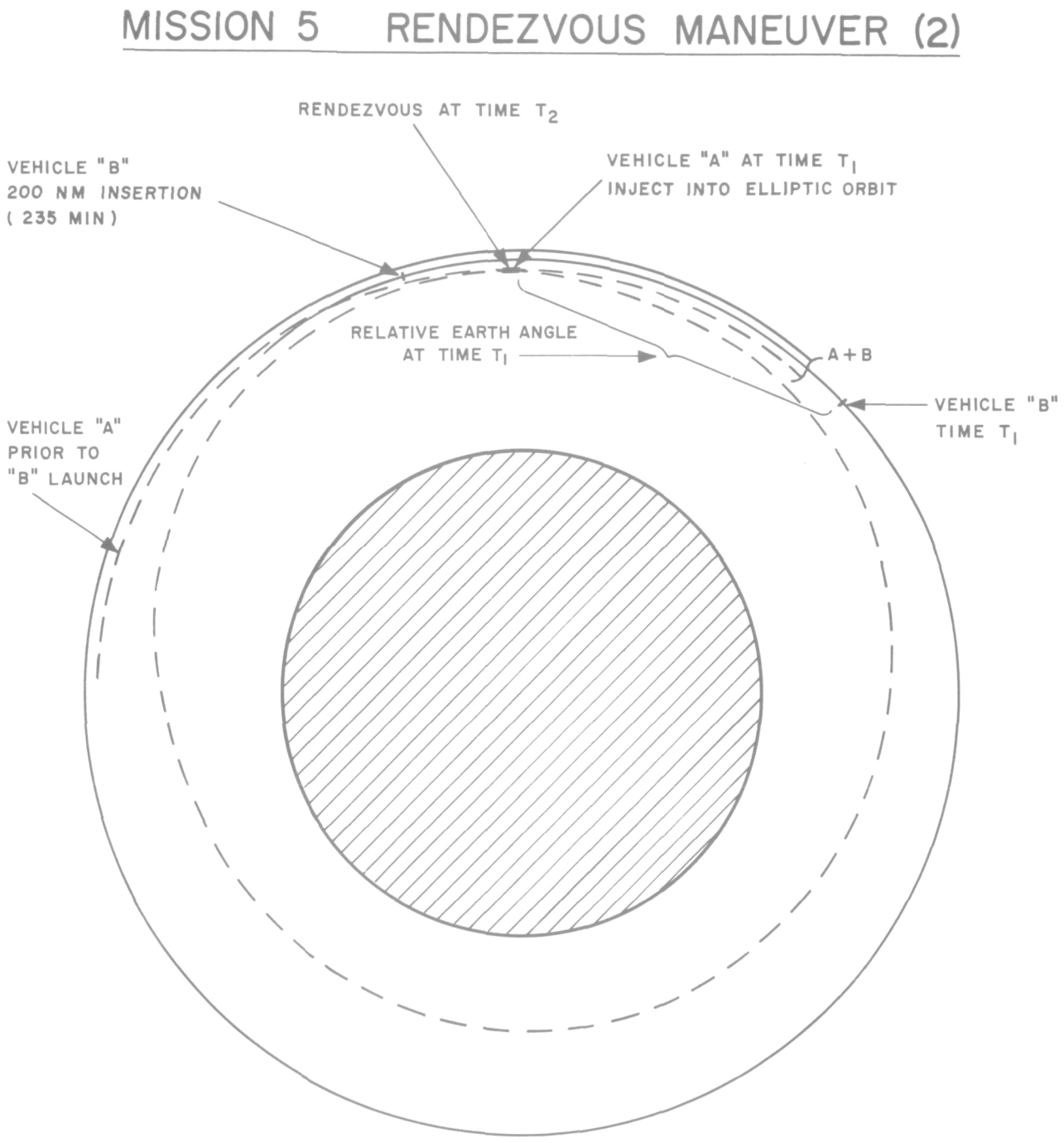

Figure 5-18. Rendezvous Maneuver (2)

orbit with an apogee of about 600 n. mi., and a perigee of 200 n. mi. The period of the ellipse is greater than the period of a 200-n. mi. circular orbit so that Vehicle B overtakes Vehicle A. The ellipse is such that rendezvous occurs at perigee after Vehicle A has spent an integral number of periods in the elliptic orbit.

Mode II, for early launches of Vehicle B, is the converse of Mode 1. Vehicle A is inserted into an ellipse with a perigee of 80 n. mi., and an apogee of 200 n. mi. The period is shorter than that of a 200-n. mi. circular orbit, and hence Vehicle A overtakes Vehicle B, and rendezvous takes place at apogee.

5.2.6 ABORT ANALYSIS

5.2.6.1 Introduction

Four major abort modes are applicable to Mission 5:

a. LES abort.
b. Suborbital abort.
c. Abort to orbit.
d. De-orbit abort.

Ground rules and mode descriptions of the major abort modes are presented in the following paragraphs, and abort logic diagrams for each abort mode are included. The relation between mission phases and the abort modes available at given points are shown in Figure 5-19.

5.2.6.2 Launch Escape System Aborts

5.2.6.2.1 Ground Rules

For either flight of this mission, LES is the primary abort mode from the time of enablement during the prelaunch terminal countdown to LES jettison during the second stage thrusting period.

Launch vehicle engine shutdown must precede LES ignition for all aborts except those initiated during the first 40 seconds of flight. During the first 40 seconds, engine shutdown is inhibited by an AMR range safety constraint.

Aborts initiated below an altitude of 120,000 feet have an automatic sequencing of spacecraft events. For aborts above 120,000 feet, the astronauts are required to jettison

the LES manually, after escape motor thrust tail-off, and to orient the spacecraft to a heat-shield-forward attitude.

5.2.6.2.2 Mode Description

The flow of operational events that follow the initiation of an LES abort are shown in Figure 5-20. The events are controlled by sequential timers and barometric switches, as indicated.

At abort initiation, launch vehicle engine shutdown is commanded (except during the first 40 seconds). The launch escape and pitch control motors are then fired simultaneously. The pitch motor burns for approximately 0.5 seconds to provide lateral separation from the booster; the escape motor thrusts for about 3.5 seconds to provide longitudinal separation.

At 11 seconds after abort, the canards are deployed. At 14 seconds after abort, if a 25,000-foot altitude barometric switch (closed prelaunch) is still closed, the tower jettison motor separates the LES (including the CM apex cover) from the CM. If the switch is open, LES jettison is delayed until reaching 25,000 feet on descent when the barometric switch is closed again. Drogue parachute deployment occurs two seconds after LES jettison. Main parachute deployment is controlled by a timer and barometric switch similar to LES jettison. At 12 seconds after drogue deployment, if a 12,000-foot altitude barometric switch is still closed, the drogue chutes are disconnected and the main chutes are deployed. If the switch is open, the drogue chutes are retained until reaching 12,000 feet on descent when the switch is closed again.

Above approximately 120,000 feet, the canard subsystem is ineffective and the pilot must manually orient the CM in a heat-shield-forward attitude. To do so, the LES must be manually jettisoned by the pilot after launch escape motor thrust tail-off (about 8 seconds). The CM apex cover is left on. Automatic RCS rate damping begins at tower jettison. At 24,000 feet, on descent, the barometric switch closes and the apex cover is jettisoned. Two seconds later the drogue parachutes are deployed. At 12,000 feet, another barometric switch closes, which disconnects the drogue chutes and deploys the main chutes.

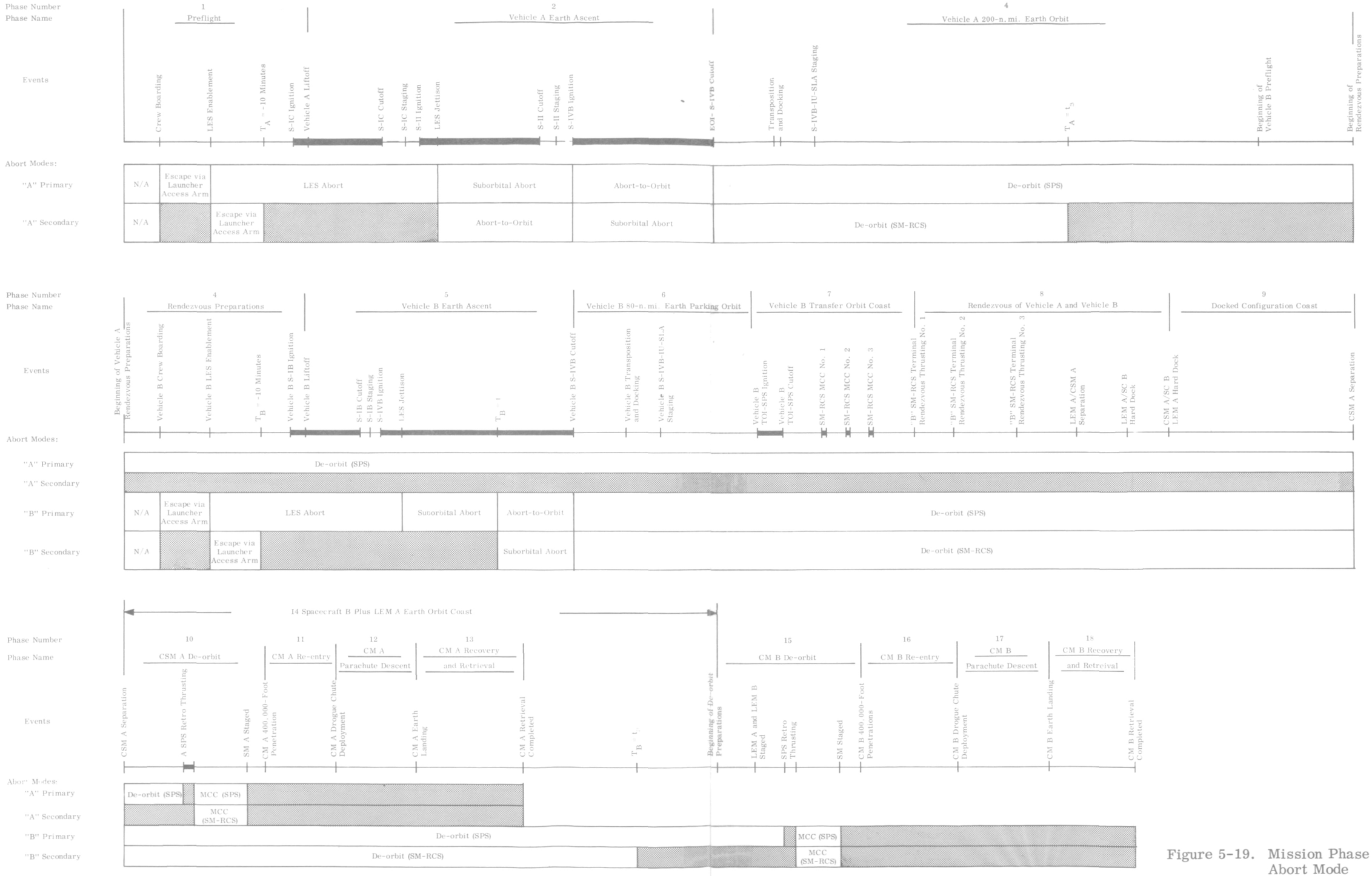

Figure 5-19. Mission Phase versus Abort Mode

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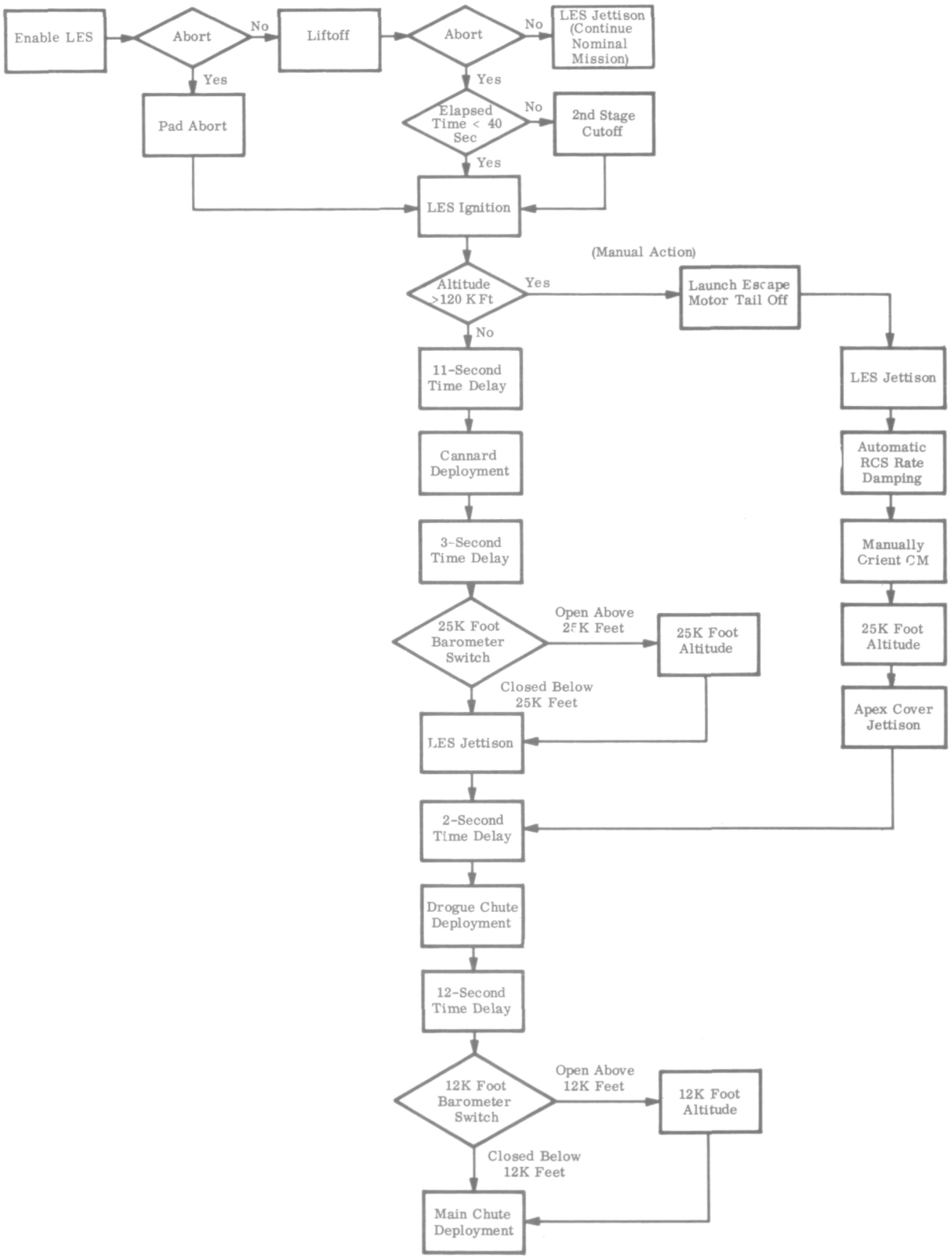

Figure 5-20. LES Abort

5.2.6.3 Suborbital Aborts

5.2.6.3.1 Ground Rules

A suborbital abort is the primary abort mode:

a. For Vehicle A, from LES jettison to S-IVB ignition.

b. For Vehicle B, from LES jettison to t_1 seconds after liftoff.

In this abort mode, the CM lands in the AMR. After S-IVB ignition, or t_1 seconds, a suborbital abort is the secondary abort mode, in which the CM lands in the Indian Ocean recovery area.

Launch vehicle shutdown must precede the initiation of spacecraft abort action.

The SM must be separated from the CM at some fixed increment of time after SM engine shutdown following completion of the suborbital abort maneuvers. The CSM abort separation procedure will utilize both an SM RCS and an SPS burn of a fixed initiation time, ΔV and attitude.

Suborbital aborts before S-IVB ignition (primary abort mode) require that the CM fly a maximum lift trajectory during entry to avoid excessive decelerations. Suborbital aborts after S-IVB ignition (secondary abort mode) require SPS posigrade thrust to obtain the range of the landing site. A constant bank angle of approximately 60 degrees will be flown by the CM during entry.

5.2.6.3.2 Mode Description

A detailed sequence of events during a suborbital abort is shown for Vehicle A in Figure 5-21 and for Vehicle B in Figure 5-22. Launch engine shutdown precedes the initiation of a suborbital abort, after which the SM RCS is used to separate the CSM and the launch vehicle in order to clear the SPS engine nozzle of the Spacecraft LEM Adapter (SLA). The SPS engine nozzle must be clear of the SLA, by about 45 inches, before the SPS can be ignited. It may be necessary to fire the SPS as soon as possible to damp out pitch and yaw rates that could exist at abort because of a launch vehicle control system failure. After the SM RCS separation and SPS damping maneuvers are completed, a turnaround to the required thrust attitude is achieved by pulsing the SM RCS.

Depending on the range control required, the SPS can be burned for variable lengths of time in either a posigrade or retrograde attitude. The time delay to thrust initiation can

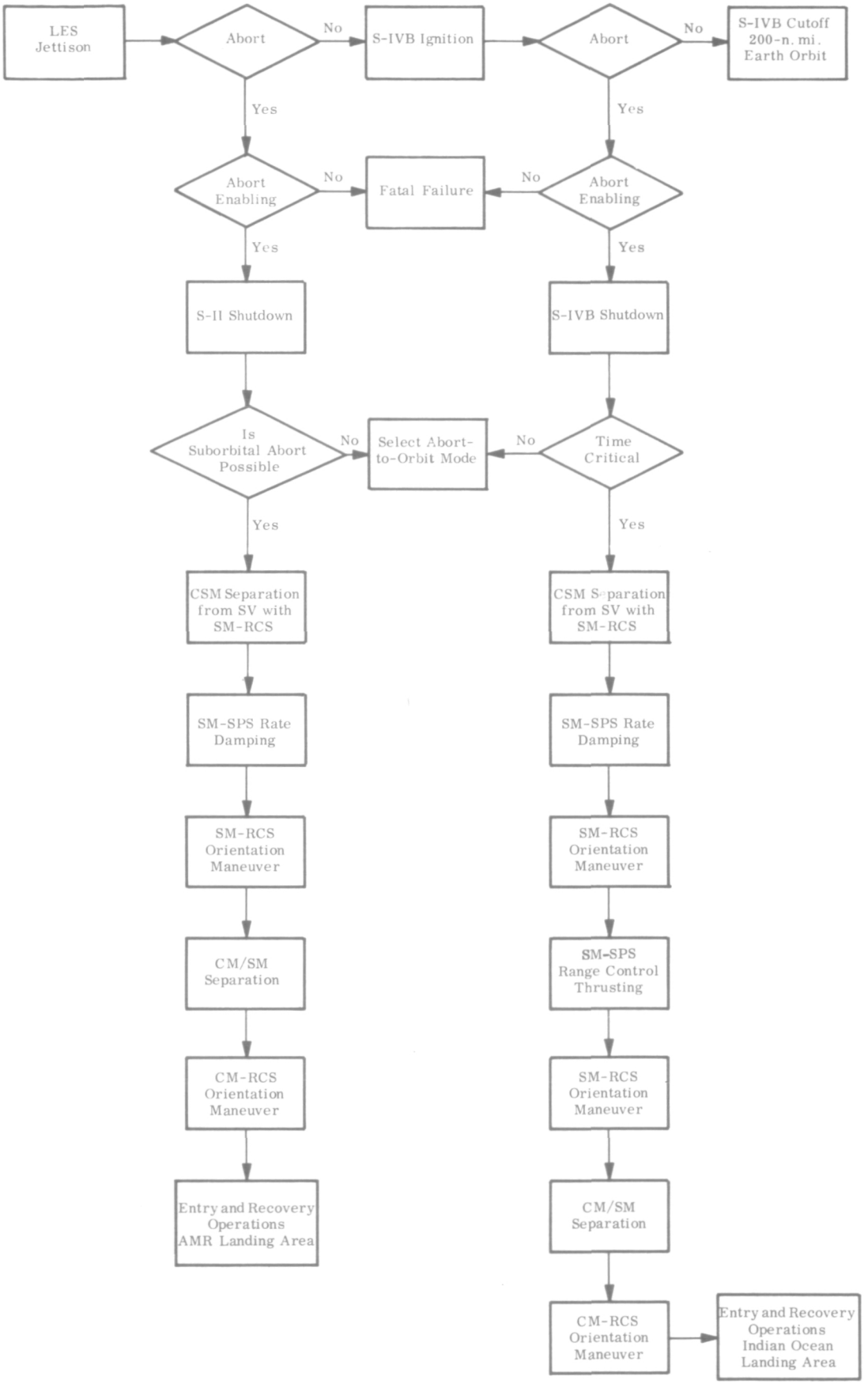

Figure 5-21. Suborbital Abort Vehicle A

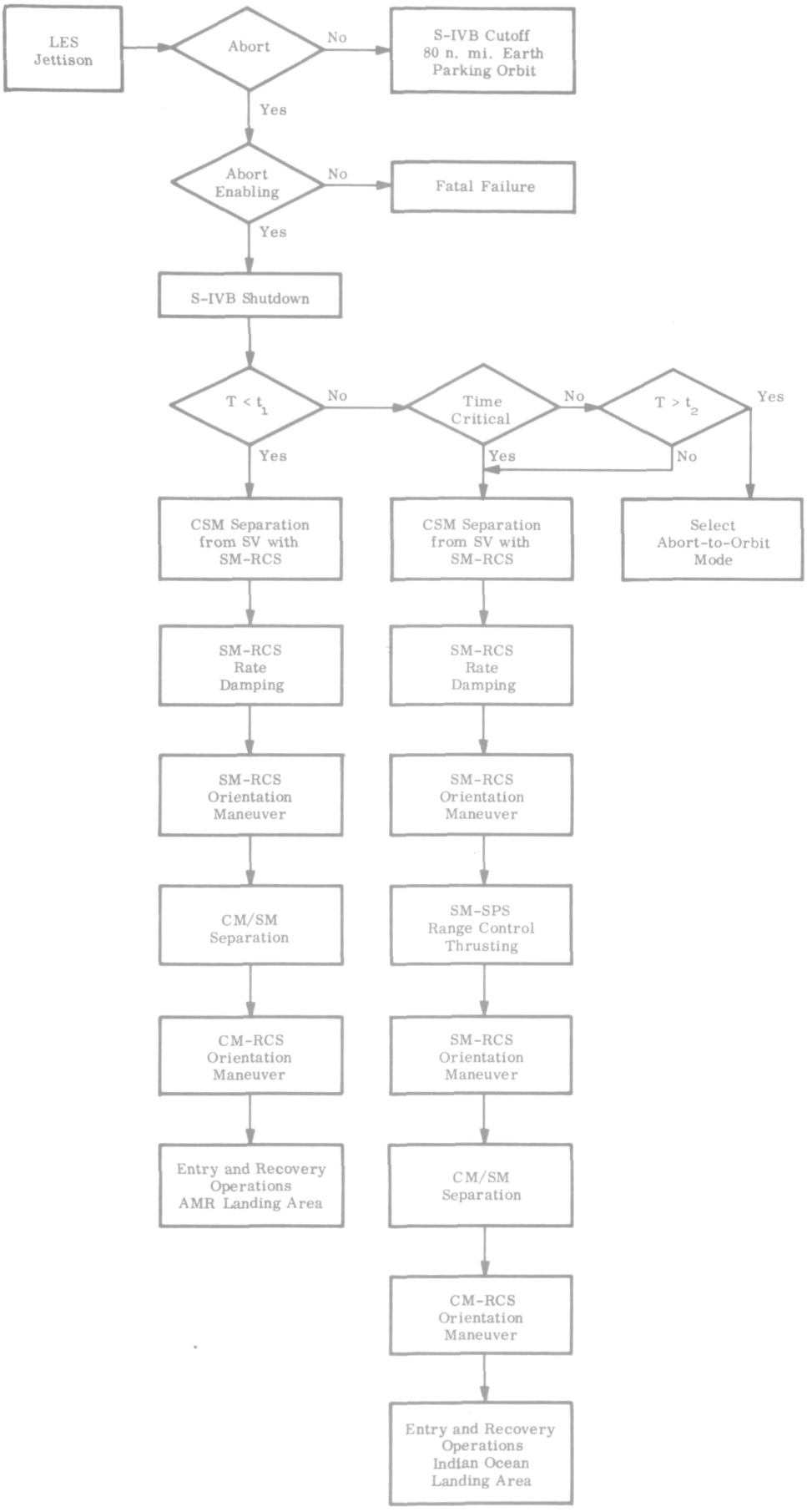

Figure 5-22. Suborbital Abort Vehicle B

be varied. After the separation from the SM, the CM is capable of re-entering ballistically or with variable lift.

SPS range control thrusting is not used for suborbital aborts in the primary mode. The CM, flying a maximum-lift trajectory, will land within the confines of the AMR. The SM RCS establishes the correct CM/SM separation attitude, after SM RCS launch vehicle separation and SPS damping maneuvers. The CM has no translatory thrust capability. Following separation, the CM RCS is used to orient the CM for a maximum-lift entry.

During an abort in the secondary mode, SPS posigrade range control thrusting is utilized to avoid landing on the African continent and to ensure a CM landing in the Indian Ocean recovery area. Following SPS range control thrusting, the SM RCS establishes the correct CM/SM separation attitude and then the separation velocity. Following separation, the CM RCS orients the CM for a constant bank angle entry utilizing an approximate half-lift capability.

5.2.6.4 Aborts to Orbit

5.2.6.4.1 Ground Rules

The advisability of aborting to orbit, once the capability exists, depends on the malfunction. If crew safety is not diminished, aborting to orbit has the advantage of greater flexibility in landing site selection. Ground rules for aborts to orbit are as follows:

a. Vehicle A - An abort to orbit is the primary abort mode from the ignition of the S-IVB stage to S-IVB cutoff, and the secondary abort mode from LES jettison to S-IVB ignition. The abort utilizes the SPS to attain orbit. All time critical aborts will be suborbital. Launch vehicle shutdown must precede initiation of spacecraft abort action.

   The S-IVB or SM propulsion and guidance systems can be used for the abort-to-orbit abort mode. For those cases where SPS thrusting is required, the CSM abort separation procedure will utilize both an SM RCS and an SPS burn of fixed initiation time, ΔV and attitude. Aborts-to-orbit require a fixed amount of SM propellant for de-orbit capability.

Aborts-to-orbit prior to S-IV ignition (secondary abort mode) will be dependent on the following time constraints:

(1) Aborts initiated prior to T +140 seconds will utilize the S-IVB propulsion plus the SM propulsion.

(2) Aborts initiated after T +410 seconds but prior to T +490 seconds will utilize the S-IVB propulsion.

(3) Aborts initiated after T +490 seconds will utilize the SPS.

b. Vehicle B - An abort-to-orbit is the primary abort mode from t_1 seconds after liftoff to S-IVB cutoff at parking orbit insertion. S-IVB shutdown must precede initiation of spacecraft abort action.

All time critical aborts will be suborbital.

The CSM separation procedure for an abort-to-orbit will utilize both an SM RCS and an SPS burn of fixed initiation time, ΔV and attitude.

5.2.6.4.2 Mode Description

A detailed sequence of events during aborts to orbit is shown for Vehicle A in Figure 5-23; for Vehicle B, in Figure 5-24. The mode descriptions for the two flights are as follows:

a. Vehicle A - The S-IVB and/or SPS provide the necessary propulsive capability to place the spacecraft into a nominal 200-n. mi. circular earth orbit and are used as dictated by the time of abort, the capability of each vehicle, and the propellant that must be reserved for de-orbiting.

The SM-SPS can supply the necessary ΔV to place the CSM in the 200-n. mi. earth orbit from an abort occurring ( ) seconds after liftoff. An SPS abort-to-orbit from an abort at T +( ) seconds is accomplished by thrusting the SPS for approximately ( ) seconds. The S-IVB can supply the necessary ΔV to place the CSM in the 200-n. mi. earth orbit from an orbit occurring ( ) seconds after liftoff. An S-IVB abort-to-orbit from an abort occurring at T +( ) seconds is accomplished by burning the S-IVB for approximately ( ) seconds. Both the S-IVB and SPS must be used for an abort-to-orbit for an abort occurring between T +( ) seconds to

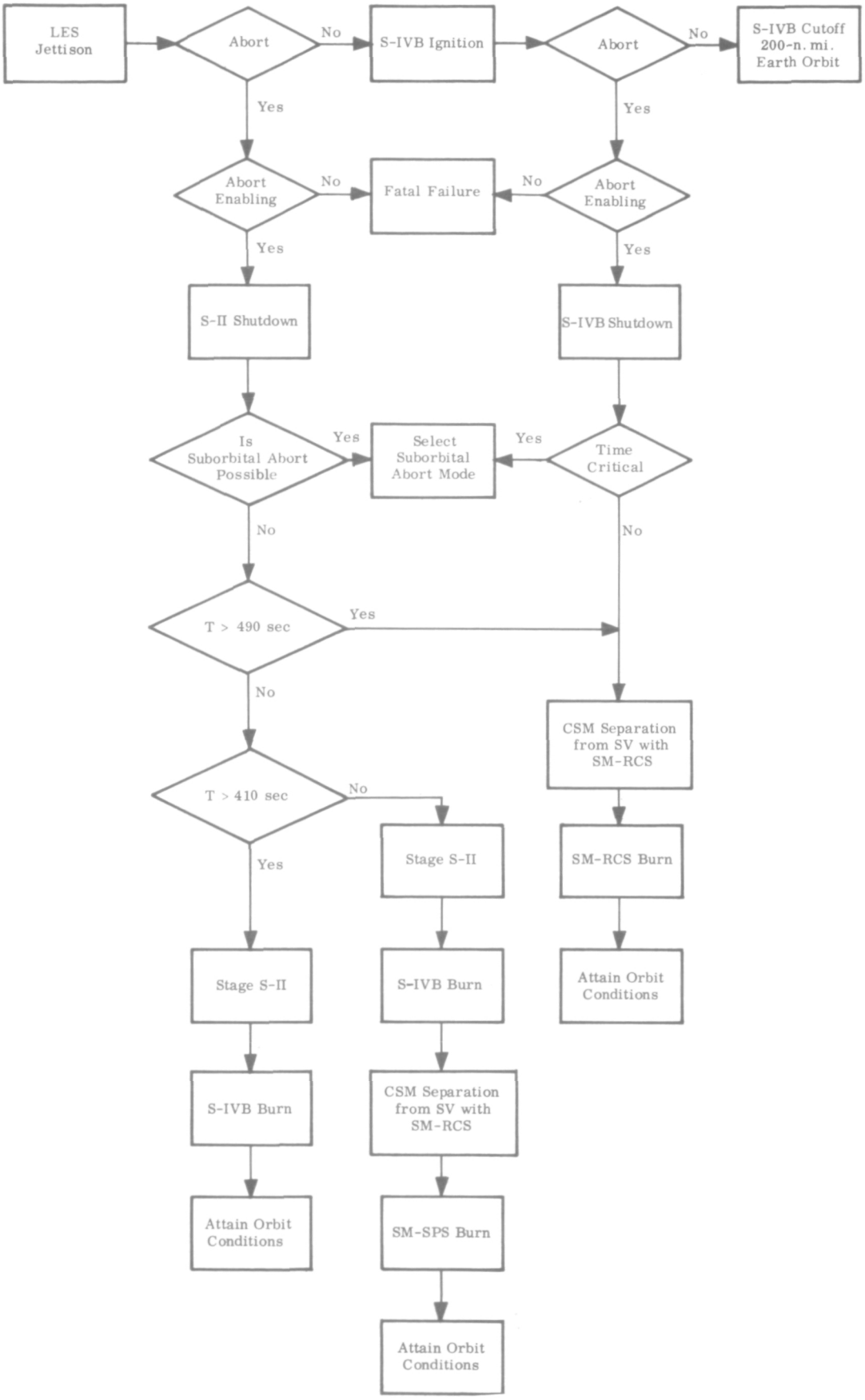

Figure 5-23. Abort-to-Orbit Vehicle A

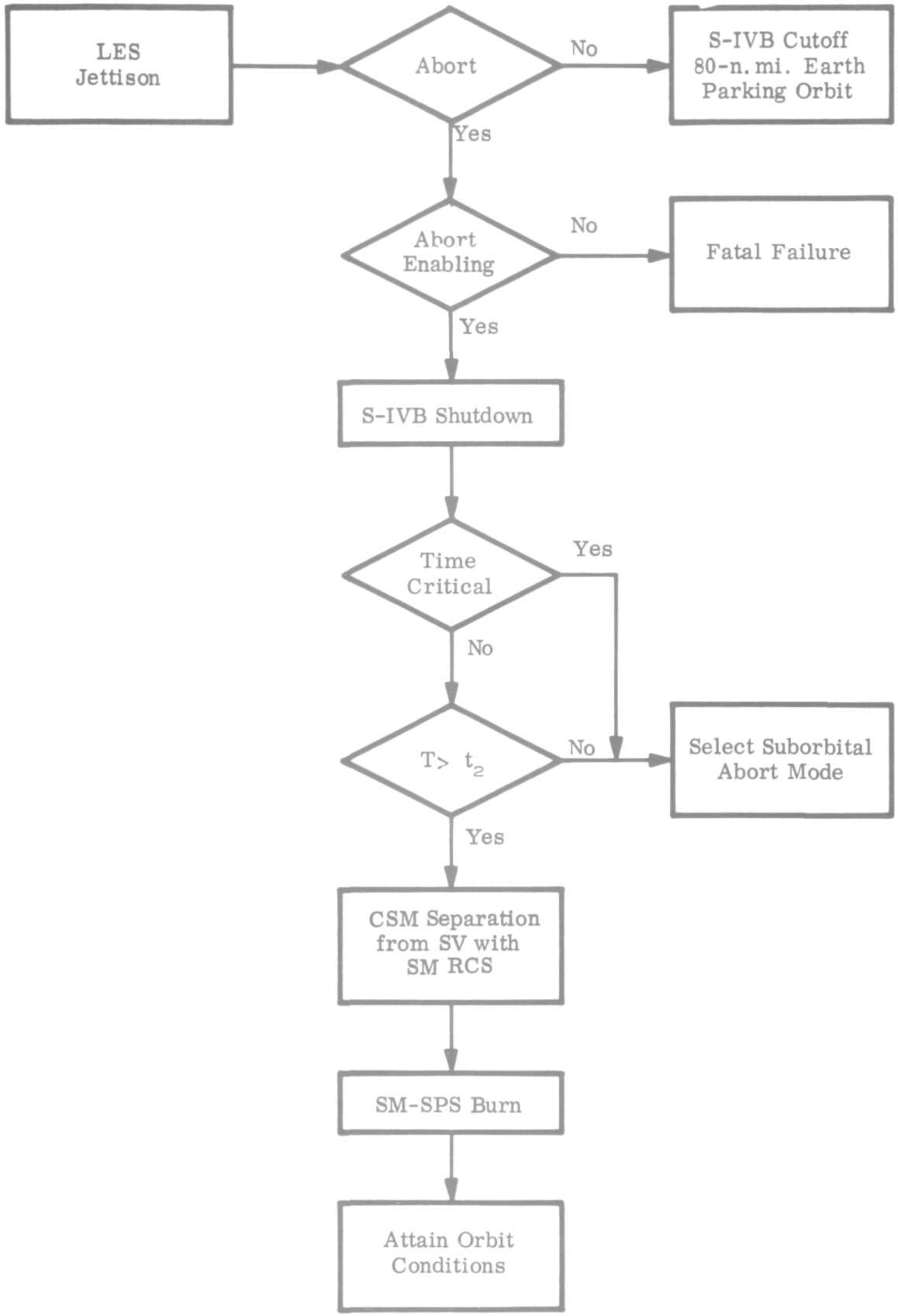

Figure 5-24. Abort-to-Orbit Vehicle B

T +( ) seconds. The abort procedure requires an S-IVB thrust-to-burnout and SPS thrusting to complete the orbit insertion maneuver.

An abort-to-orbit could be to any altitude between 100 and 200 n. mi., the lower limit determined by the orbital lifetime. An abort to an orbital altitude less than the nominal 200 n. mi., would permit the abort-to-orbit mode to be utilized at an earlier mission time, and permit an abort requiring less ΔV. In a lower orbit, the time for de-orbit and recovery would be reduced.

b. Vehicle B - An abort-to-orbit is the primary abort mode from T = t_1 to S-IVB cutoff; however, all time critical aborts dictate the selection of the suborbital abort mode, the secondary abort mode for this period.

Following booster shutdown, the SM RCS is used to separate the CSM and the launch vehicle to clear the SPS engine nozzle of the SLA by about 45 inches so that the SPS can be ignited. Following CSM separation, the SPS is ignited and thrusted until orbit insertion is achieved.

Sufficient SPS propellant must be reserved for the de-orbit maneuver. An SPS abort-to-orbit from an abort occurring at T = t_1 requires approximately ( ) seconds of SPS thrusting.

5.2.6.5 De-Orbit Aborts

5.2.6.5.1 Ground Rules

The only abort possible for either flight, from S-IVB cutoff at the time of earth parking orbit insertion, to the ignition of the SPS for the nominal de-orbit maneuver, is an early de-orbit maneuver. The primary abort mode utilizes the SPS to accomplish the de-orbit maneuver; the secondary abort mode utilizes the SM RCS. The secondary abort mode is not available after T = t_3.

The spacecraft must consist of only a CSM manned by the three-man crew, at the time of initiation of de-orbit thrusting. The de-orbit maneuver will utilize a single propulsive burn at a fixed retrograde attitude and for a specified ΔV. The CM/SM separation procedure will utilize a SM RCS translation and roll maneuver.

When time permits, the de-orbit maneuver will be calculated to permit entry to one of the planned recovery areas, preferably to the designated primary recovery area.

5.2.6.5.2 Mode Description - Vehicle A

The primary de-orbit abort mode utilizes the SPS for the de-orbit maneuver; the secondary de-orbit abort mode utilizes the SM RCS to provide the de-orbit propulsion. However, the secondary abort mode is only available until $T = t_1$, the mission time at which available SM RCS propellant is less than that required for the de-orbit maneuver. A detailed sequence of events during a de-orbit abort is shown in Figure 5-25.

If abort action is required prior to S-IVB staging following transposition and docking, the abort procedure is to stage the S-IVB-IU-SLA-LEM, use the SM RCS to properly orient the CSM, and de-orbit by an SPS retro burn.

If abort action is required after S-IVB staging following transposition and docking, the abort procedure is to stage the LEM, use the SM RCS to properly orient the CSM, and de-orbit by an SPS retro burn.

If CSM A abort action is required during the period from CSM A/LEM A separation for rendezvous with Vehicle B to CSM A nominal redocking with LEM A, the abort procedure requires a CSM A/LEM A redocking, the transfer of the LEM A crew to CSM A, and LEM A/CSM A separation before CSM A de-orbit abort thrusting. Either the CSM A or LEM A can be the active vehicle in the redocking operation, but ordinarily the LEM A is active.

If CSM A abort action is required during the docked configuration coast, the abort procedure requires that CSM A be manned by a three-man crew before the CSM A is separated from the Vehicle B/LEM A configuration. Following separation, the de-orbit abort procedure is as described above.

5.2.6.5.3 Mode Description - Vehicle B

A detailed sequence of events during a de-orbit abort for Vehicle B is shown in Figure 5-26. If a de-orbit abort is required before the completion of transposition and docking, the S-IVB-IU-SLA-LEM configuration is staged. If transposition and docking is complete, only the LEM is staged. Following staging operations, the CSM is properly oriented for the de-orbit maneuver by means of the SM RCS.

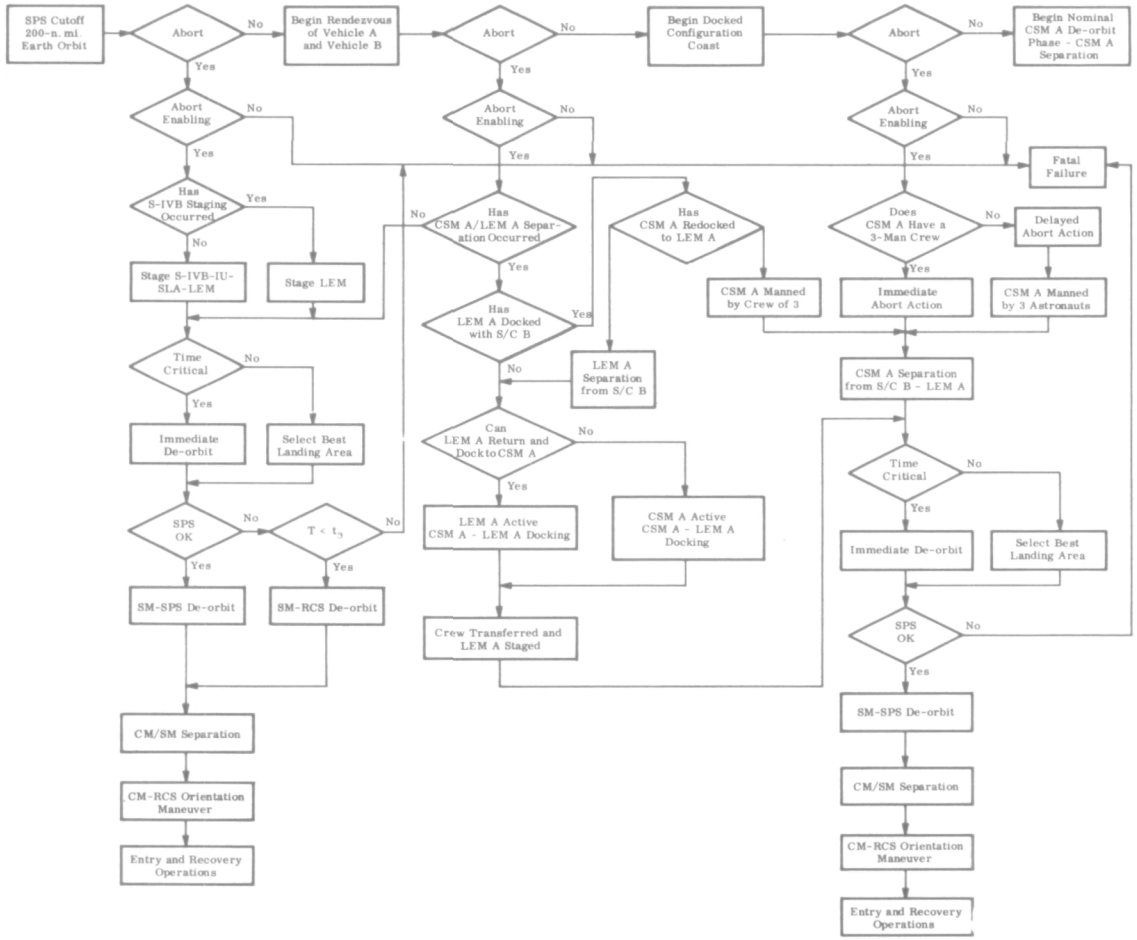

Figure 5-25. De-orbit Aborts Vehicle A

The SPS is the primary propulsive mode for the de-orbit abort. The SM RCS serves as a secondary propulsive mode until a mission time of t_2, the time at which the SM RCS propellant usage has depleted the ΔV capability to an amount insufficient for the required de-orbit thrusting.

The de-orbit abort procedure requires that the SPS be shutdown if the abort should occur during any period of SPS thrusting.

The sequence of events following the de-orbit abort thrusting maneuver is the same as the events that follow the nominal de-orbit maneuver.

If CSM B abort action is required during the period between LEM A docking to LEM B and nominal CSM A separation, the de-orbit abort procedure requires that the CSM B be manned by the three-man crew before CSM B is separated from the LEM A/LEM B or CSM A/LEM A/LEM B configuration. Following separation the de-orbit abort procedure is as described previously.

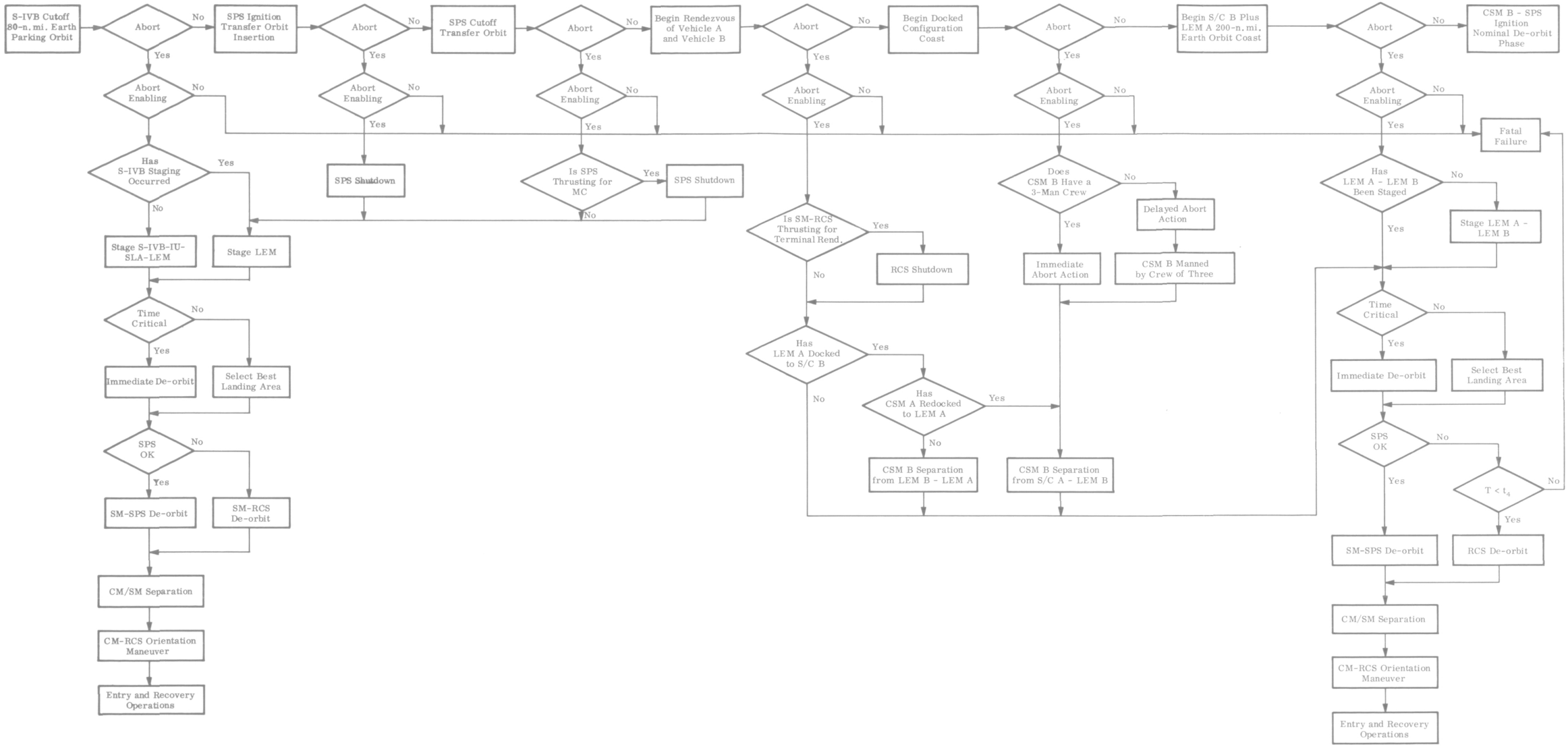

Figure 5-26. De-orbit Aborts Vehicle B

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